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STUDY OF NAVIGATION AND GUIDANCE OF LAUNCH VEHICLES HAVING CRUISE CAPABILITY
FINAL REPORT, VOLUME 3 OF 4
ALTERNATE NAVIGATION-GUIDANCE CONCEPTS
PHASE I

by

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FOREWARD

This document reports on an investigation by The Boeing Company from June 10, 1966 to March 10, 1967 of the navigation and guidance of a two stage launch vehicle (hypersonic stage 1/rocket stage 2) under contract NAS 2-3691. The Technical Monitor for the study was Mr. Hubert Drake of the NASA Mission Analysis Division, Moffett Field, California with Co-monitor Mr. Frank Carroll of the NASA Electronics Research Center, Cambridge, Massachusetts.

The Final Report is prepared in four volumes:

Volume 1 - Summary Report, Boeing Document D2-113016-4

Volume 2 - Trajectory Parametric and Optimization Studies, D2-113016-5

Volume 3 - Alternate Navigation-Guidance Concepts (Phase I), D2-113016-6

Volume 4 - Detailed Study of Two Selected Navigation-Guidance Concepts (Phase II), D2-113016-7.

Boeing personnel who participated in the study reported in this volume (Volume 3) include J. A. Retka, program manager; C. R. Glatt, payload performance analysis; D. Harder, guidance concepts; T. Seavoy, navigation techniques; N. E. McAdory, reliability; and D. Minden, parametric cost data. Assistance and consultation was also provided by S. Augustyniewicz, M. Mobley, G. Yamamoto, and H. Donnel.

ABSTRACT

This report details the work done on Phase I of Contract NAS 2-3691, A Research Study of Navigation and Guidance of Launch Vehicles Having Cruise Capability. The Phase I study examines the navigation and guidance requirements of a two stage launch vehicle, consisting of an airbreathing recoverable first stage and a rocket powered second stage. The effects of mission constraints and vehicle characteristics on guidance requirements are determined. Error analyses of selected guidance and navigation systems are performed. The effect of the various guidance and navigation systems chosen for study on payload, reliability, cost, and safety is used in system trade studies. Two candidate navigation and guidance systems are chosen for further detailed study.

KEY WORDS

Recoverable Booster
Navigation
Guidance
Flight Performance
Trade Studies

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REFERENCES

1. Monthly Progress Report No. 2, Contract NAS 2-3691, A Research Study of Navigation and Guidance of Launch Vehicles Having Cruise Capability, July 11 to August 10, 1966.

2. "Study of Filter Techniques in Distant Planet Orbit Parameter Estimation", D. Harder and D. E. Conner, AIAA/JACC Guidance and Control Conference, Seattle, Washington, August 15-17, 1966.

3. Tibbals, M. L., OMEGA Navigation - Aeronautics and Astronautic Applications, AIAA Paper 64-419, 1st AIAA Annual Meeting, Washington, D. C., June 21, 1964.

4. OMEGA - A World-Wide Navigation System, System Specification and Implementation, by J. A Pierce, W. Palmer, A. D. Watt, and R. H. Woodward, published by Packard and Burns Electronics, P&B Publication No. 886, 1 June, 1964.

5. Baltzer, O. J., Aircraft Navigation and Position Determination by Measurements of VLF/OMEGA Doppler Rates, IEEE Twelfth Annual Conference on Aerospace and Navigational Electronics, Baltimore, Maryland, Oct. 27-29, 1965.

6. Kershner, R. B., Status of the Navy Navigation Satellite System, Presented to the American Astronautical Society National Meeting on Practical Space Applications, San Diego, California, February 21, 1966.

7. Anderson, R. E., A Navigation System Using Range Measurements from Satellites with Cooperating Ground Stations, Navigation, V. .11, No. 3, Autumn 1966, P. 315.

8. Keats, E. S., A Navigation System Using Distance and Direction Measurements From a Satellite, Navigation Vol. 11, No. 3, Autumn 1964, P. 335.

9. Breckman, J., A Public NavSat System, Space/Aeronautics, June 1966, P. 92.

10. White, R. N., Airborne Doppler Radar Navigation of Jet Transport Aircraft, IRE Transactions on Aerospace and Navigational Electronics.

11. C5A Guidance Subsystem Proposal, Boeing Document D2-11056-1, April 20, 1964.

12. Blumhagen, V. A., Stellar-Inertial Navigation Systems for Cruise Vehicles, Autonetics Publication 552-A-26, April 1963.

13. Christensen, T. W., Advanced Development of ESG Strapdown Navigation Systems, IEEE Transactions on Aerospace and Electronic Systems, Vol. AES-2, No. 2, March, 1966, P. 143.

14. A. A. Morgan and G. W. Braudaway, An Explicit Multi-Stage Boost Guidance for Minimum Fuel Consumption.

1.0 Introduction and Summary

The study is directed at determining the feasibility, capabilities, and limitations of navigation and guidance systems for a two stage launch vehicle having an aerodynamic, air breathing first stage and a rocket second stage. The basic mission is to fly a 3704 Km (2000 nautical mile) offset distance to the orbital plane of a satellite, turn into the plane and separate the second stage which then accomplishes rendezvous of the payload with a target satellite. The first stage then returns to its base. Phase I, the first four months of the nine month study, is a comparative analysis of alternate navigation and guidance concepts. This Volume of the Final Report covers the navigation-guidance work accomplished during Phase I and supports the recommendation of two navigation and guidance concepts for detailed study during Phase II. This Volume is essentially the same as the Interim Report, D2-113016-3, with some revisions and with the use of the International System of Units. D2-113016-4 is a Summary Report; D2-113016-5 presents the nominal trajectory studies and the trajectory optimization results, and D2-113016-7 presents the detailed studies of the two selected navigation-guidance concepts.

The overall objective of the study is to determine if substantial improvements in navigation and guidance technology are required in order to avoid significant losses in mission performance with this launch vehicle. A rescue mission is a typical rendezvous mission that the launch vehicle is required to perform. The vehicle characteristics and initial estimate to flight performance capabilities are described. The nominal flight profile is developed in D2-113016-5.

The alternate navigation concepts that have been considered for Stage 1 are doppler radar, astrocompasses, inertial navigators, doppler inertial, stellar inertial, navigation satellites and the Omega radio navigation systems. Alignment of a Stage 2 platform from a Stage 1 master platform has been studied. The accuracy performance of the guidance system is described by the vector velocity error magnitude at the point of injection into the rendezvous transfer orbit.

The critical guidance problem is the correction of time errors that are caused by off-nominal environment or vehicle characteristics. Correction methods are developed for the cruise phase, for direct ascent rendezvous, and by use of a parking orbit mode. The form of the guidance equations is outlined and the associated guidance computer requirements are determined.

Payload performance capabilities and penalties are developed for the alternate guidance and navigation concepts. The payload weight panalties for correcting time errors are given for alternate correction methods. Direct ascent and parking orbit performance penalties are compared. It has been demonstrated that a combination of first stage and second stage correction modes are feasible for compensating for expected off-nominal conditions with acceptably small payload performance penalties.

The payload performance penalties to correct navigation errors are also developed and compared. The performance penalties are incurred in the terminal phase of the rendezvous mission and on the base return phase for Stage 1. A target seeking radar measures the navigation errors so that corrections can be made during the terminal phase of rendezvous. Short range radio aids are used during the stage landing field approach. A wide range of accuracy performance has been studied for both Stage 1 and Stage 2 navigation systems.

Comparative reliability analyses and Cost Analyses have been made. An effort has been made to identify a number of the factors that cause a wide range of values for both reliability and cost estimates. The estimates made represent current data. It should be recognized that both reliability and cost of electronic components and computers have been changing rapidly with time. The data are believed to be fairly accurate for relative comparisons and are useful for performance trades to determine the point of diminishing returns.

The principal conclusions of the Phase I study are:

* Navigation and guidance for the rendezvous mission is feasible with state of the art technology.

* The apparent optimum state of the art navigation and guidance system in terms of relative performance versus cost consists of a medium accuracy inertial system (.01°/hour gyro drift) on Stage 2; a 18.5 km/hour (ten nautical miles/hour) inertial system on Stage 1; and medium capability digital computers on each stage. However, consideration of safety, air traffic control, collision avoidance, refueling rendezvous, relative development effort, and other current navigation applications support the recommendation of a 1.85 km/hour (one nautical mile/hour) accuracy class navigator for Stage 1.

* With advancing computer technology, the cost of modal flexibility and optimized data filtering is nominal in terms of added computer requirements. Therefore, in the study of advanced concepts, primary emphasis should be given to the study of optimal data filtering techniques, and to the study of the organization of the sensors and computers in a system with maximum function-level redundancy.

Review of the Phase I results by the NASA Mission Analysis Division, NASA Electronic Research Center, NASA Langley Research Center and others resulted in a request for additional Phase II effort on trajectory optimization and the determination of feasibility of the lambda matrix guidance technique or an equivalent modern control theory approach for the rendezvous problem.

2.0 Analyses of Alternate Navigation and Guidance Concepts

2.1 Study Objective, Mission Goals and Constraints

2.1.1 Introduction, Statement of Work

"Various advanced launch vehicle concepts are being studied having potential usefullness in providing transportation to and from orbit. A number of studies have been conducted by government agencies and contractors, of launch vehicle systems having varying degrees of aerodynamic flight capability. The horizontal take-off and horizontal-landing vehicle studies have included "Recoverable Boosters" (USAF), "Aerospace-plane" (USAF), and "10-Ton Orbital Transport" NASA. These various studies have noted the potential ability of air breathing first stages to perform a wide variety of missions, particularly the offset missions. An "offset" mission occurs where the launch vehicle accomplishes a significant lateral displacement in aerodynamically supported flight, possibly including some loiter, and then in the proper orbit plane, proceeds to establish the launch conditions required by the upper stage. This capability is of particular benefit for missions involving rendezvous or near approach to some object in orbit. A limitation of all these studies, however, has been the assumption that the guidance and navigation problem is trivial and that the equipment required is of negligible weight and volume, and will be readily available. However, for the rendezvous task the navigation and guidance system requirements may be difficult to meet unless substantial improvements in technology are incorporated or significant losses in mission performance are tolerated. A detailed study is now required to determine the required functions and capabilities of the on-board guidance and navigation equipment, its probable physical characteristics, and also to examine the mission performance losses associated with existing and improved navigation and guidance system technology. The results of this study will enable an evaluation of the effects that the navigation and guidance system will have on the overall vehicle design and mission performance and will aid in identifying critical research areas and areas in which increased research might result in significant system improvement."

2.1.2 Objectives

The objectives of this study are: (1) To study guidance schemes for a two-stage earth-to-orbit vehicle having an aerodynamically supported, air-breathing, first stage and a rocket powered second stage; (2) to examine means of operating the vehicle which minimize mission performance losses and/or reduce complexity in the navigation and guidance system; (3) to select two promising schemes and define them in detail including sensors, computers, command systems and associated on-board equipment; (4) to perform sensitivity studies on the selected system particularly with regard to accuracy capability, safety, mission reliability;

(5) to investigate the application of elements of this system to a cruise mission, and (6) to assess probability of successful development and define critical research areas associated with the development of the navigation and guidance system.

Additional study objectives are: (7) Consider both near term and advanced technology concepts to provide direction to technology research required to implement the recoverable launch vehicle concept, and (8) Provide a maximum navigation and guidance system flexibility to maximize the potential applications of a recoverable launch vehicle. The cost of obtaining the flexibility should be determined.

2.1.3 Study Constraints

The following ground rules apply:

(1) The baseline vehicle system is a two stage-to-orbit system described in Section 2.2. The nominal missions are also defined in Section 2.2. Note that both manned and unmanned second stages are to be considered. A rescue mission is a typical manned mission to be considered.

(2) Take-off is not restricted to any geographical location and the take off runway is assumed to be unaligned with the ascent trajectory.

(3) All flight phases subsequent to take off are unassisted by ground tracking information although radio navigation aids are acceptable. World-wide operation is required.

(4) The accuracy required of the air-breathing stage is primarily associated with the target staging conditions. These conditions in turn are established by the requirement that the ballistic stage must achieve injection into the target orbit with accuracy compatible with rendezvous requirements, which for the present study will be taken as equivalent to that of Gemini. The target orbit is 485 km (262 nautical miles) altitude. The study constraint that Gemini rendezvous requirements be satisfied applies to target seeker range for cooperative targets and the fuel budget for correction of errors. Terminal maneuver requirements for special military missions or for the detection of small targets are payload design considerations that can be considered outside the scope of the study.

(5) The target station orbit and location shall be considered to be adequately established at first stage take off. Corrections to orbital characteristics, however, should be accepted up to the time of staging.

(6) The launch system shall be considered to operate for 150 flight hours per year for a total life of ten years. The cruise vehicle discussed shall be considered to operate 2000 hours per year for a total life of ten years. Design life reliability, preflight and postflight checkout requirements are to be determined.

(7) The navigation and guidance system contribution to mission success shall be considered, initially, as no less than 95 per cent with a design objective of 99 per cent or better. The abort problem is considered quite important. Typical abort problems to consider for their effect on the N & G requirements are ability to safely reduce speed when required; ability to return to an emergency landing field both before and after launching the second stage; and ability to handle engine out, environmental control malfunction, or poor engine performance situations.

(8) The U. S. Standard Atmosphere - 1962 shall be used in this study.

(9) The manned payload is recoverable, the unmanned payload is not recoverable. The effect on the navigation and guidance configuration and mission costs should be considered for the two cases. It is assumed that the second stage guidance, engines, etc., will be recovered with the payload whenever possible. Detailed study of payload recovery navigation and guidance is not required.

(10) The effect of the number of launch vehicle (and payload) crew members on the navigation and guidance design is to be considered.

(11) An itemized cost analysis is desired for development and production to support cost effectiveness studies.

(12) A preliminary design level of detail is acceptable for equipment studies. Consider interfaces between subsystems such as electrical power and environmental conditioning for potential problems.

2.2 Vehicle Characteristics

Flight performance studies done in Phase I are reported in D2-113016-5. Summarizing these results, Figure 2-2-1 shows the vehicle performance for the several missions in terms of Stage 2 weight (stage payload) and/or orbital payload. The gross weight and empty weight are the specified statement of work values. Fuel and payload were then traded using appropriate exchange ratios for orbital payload as described in Appendix A1. The performance results given are the best obtained during the preliminary Phase I studies. Improved performance results were developed in Phase II and are reported in D2-113016-5. The maximum vehicle performance is determined within the following design constraints.

Stage 1 Design Constraints

Maximum Mach Number		7
Maximum Dynamic Pressure	95,700 $\frac{\text{newtons}}{\text{meter}^2}$	(2000 psf)
Maximum Propulsion System Internal Pressure	1,379,000 N/m^2	(200 psi)
Maximum Normal Load Factor		2.5
Maximum Sonic Boom Overpressure	143 N/m^2	(3 psf)

Figure 2.2-1 **SUMMARY OF MISSION WEIGHTS, PHASE I RESULTS**

	3,704 km (2000 N.M.) Offset		1,852 km (1000 N.M.) Offset		Zero Offset		9,260 km (5000 N.M. Cruise	
	kg	(lbs)	kg	(lbs)	kg	(lbs)	kg	(lbs)
Gross Take-Off	227,000	(500,000)	227,000	(500,000)	227,000	(500,000)	227,000	(500,000)
Stage 1 (Stage Weight)	170,000	(374,500)	145,000	(319,500)	142,000	(313,500)	--	--
Empty	105,200	(230,200)	105,200	(230,200)	105,200	(230,200)	131,000	(289,300)
Fuel	65,000	(144,300)	40,500	(89,300)	37,800	(83,300)	72,100	(170,050)
Stage 2 (Stage Payload)	56,900	(125,600)	81,800	(180,640)	84,500	(186,650)	18,500	(40,650)
Empty	5,960	(13,170)	8,590	(18,940)	8,870	(19,570)	--	--
Propellant	43,800	(98,710)	64,300	(141,970)	66,400	(146,690)	--	--
Payload (orbital payload)	6,220	(13,720)	8,950	(19,730)	9,250	(20,390)	--	--

2.2.1 Nominal Vehicle Characteristics

The vehicle under consideration is an earth-to-orbit launch system composed of an aerodynamically supported air-breathing first stage and a ballistic rocket second stage. The first stage structure is made up of nickel alloy heat shields and aerodynamic control surfaces, high temperature insulation and titanium load carrying structure for wing and body. The propulsion system consists of subsonic burning liquid hydrogen turboamjets with sea level static thrust of 1.22×10^6 (275,000 pounds). Thrust and specific fuel consumption data are described in reference 1. The second stage is an expandable liquid oxygen/liquid hydrogen rocket vehicle with startburn thrust-to-weight ratio of 1.5. The vacuum specific impulse is specified to be 420 sec. The structural weight is considered to be 13.3% of the propellant weight. The configuration geometry is as follows:

Stage 1

Length	87.9 meters	(288 ft)*
Body Volume (Sears-Haack Shape)	1,600 m^3	(56,600 ft^3)*
Wing Planform (edges extended to vehicle center line)	580 m^2	(6,250 ft^2)
Wing Aspect Ratio (delta planform)		1.455
Wing chord thickness ratio		.04

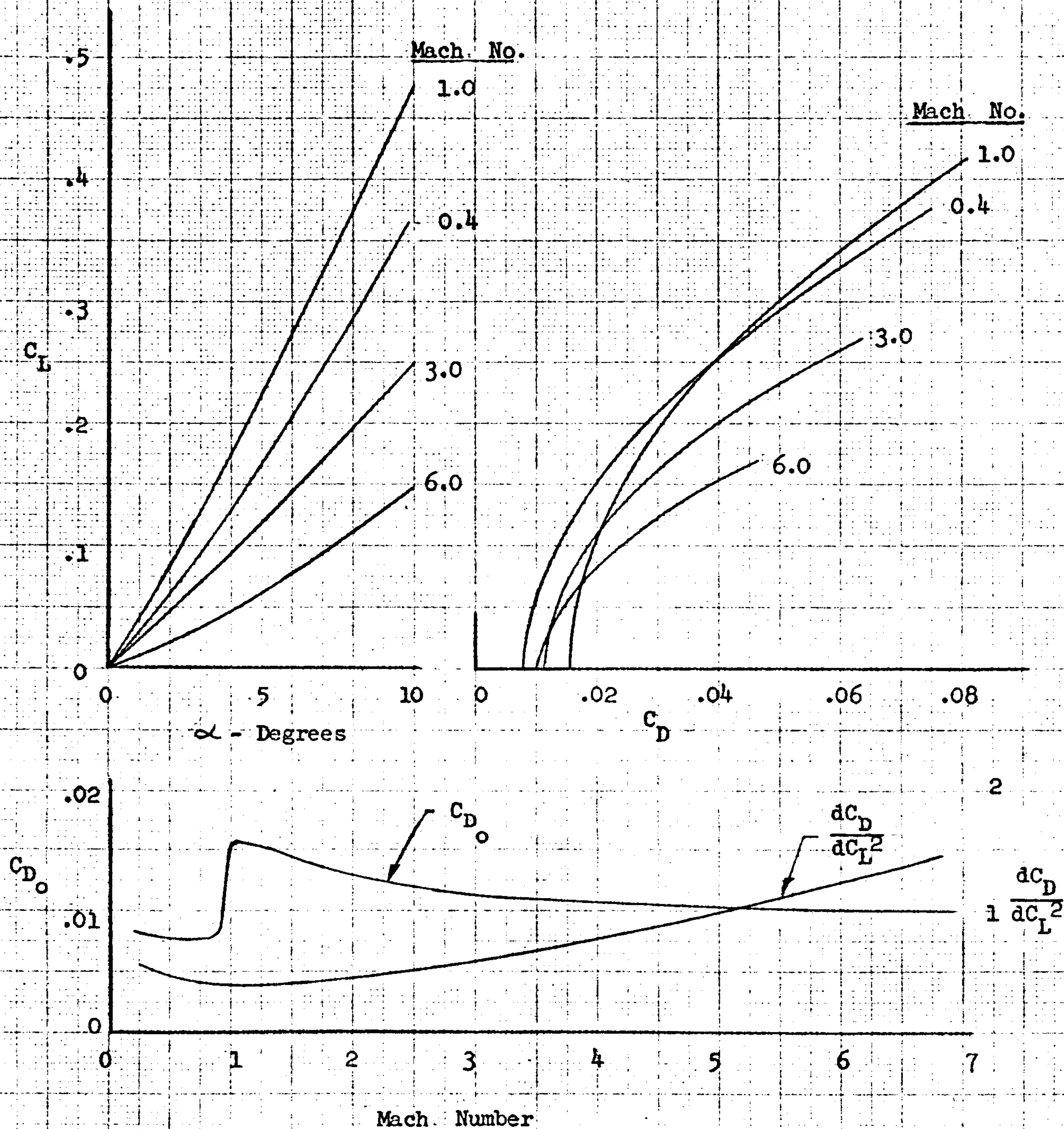
*95.4 m (312 feet), 2,020 m (71,500 ft^3) for cruise vehicle

Stage II

Length	32.6 m	(107 ft)
Diameter	2.7 m	(9.8 ft)
Reference area	5.78 m^2	(62.2 ft^2)

The aerodynamic data for the first stage vehicle is shown in Figure 2.2-2. It is based on the cruise vehicle but is considered satisfactory for the boost vehicle as well. The drag coefficient of the second stage was assumed to be a constant value of 0.13 based on the reference area of (62.2 ft^2) 5.78 m^2.

The vehicle characteristics were taken directly from the specifications for the study and supplemented where necessary with reasonable assumptions. No attempt was made to optimize or alter the prescribed data. Optimization of operational characteristics were assumed to have no feedback on vehicle characteristics.



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	Glatt	6/14/6			AERODYNAMIC CHARACTERISTICS FIRST STAGE	Figure 2.2-2
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The design weights for the 3,710 km (2000 nautical mile) offset launch mission and the 9,260 km (5000 nautical mile) cruise are as follows:

Weight	3,710 km (2000 N.M. Offset)		9,260 km (5,000 N.M. Cruise)	
Gross	227,000 kg	(500,000 lbs.)	227,000 kg	(500,000 lbs.)
Empty	104,000 kg	(230,200 lbs.)	131,000 kg	(289,300 lbs.)

Further vehicle configuration data are given in

(1) Richard H. Peterson, Thomas J. Gregory, and Cynthia L. Smith, Some Comparisons of Turbojet-Powered Hypersonic Aircraft for Cruise and Boost Missions, AIAA Paper No. 65-759, Nov. 15-18, 1965, Los Angeles, California, National Aeronautics and Space Administration, Moffett Field, California.

(2) Thomas J. Gregory, Richard H. Peterson, and John A. Wyss, "Performance Tradeoffs and Research Problems for Hypersonic Transports," Journal of Aircraft Vol. 2, No. 4, July-August, 1965.

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2.3 Alternate Navigation Techniques and Associated Error Analysis

Navigation is defined as the process of determining the current values of the vehicle position and velocity components. Alternate navigation techniques differ in the type of sensors used and in the procedures to process the measurements to minimize error effects. Stage 1 and Stage 2 navigation techniques will be described and typical accuracy performance will be estimated. Attitude alignment transfer techniques will also be described.

There are two technology bases for obtaining navigation and guidance systems for the airbreathing first stage/rocket second stage launch vehicle. The first technology base is the family of guidance systems that have been developed for ground launched missiles and launch vehicles. The second base is the family of navigation systems developed for military and commercial airplanes. The state of the art is conveniently divided into three categories: (1) existing operational systems, (2) systems under development, and (3) possible future systems.

Navigation and guidance of the air launched rocket second stage differs from conventional ground launched guidance in the greater difficulty of specifying the initial conditions at the start of rocket thrust. The initial conditions required are the three components of position, three components of velocity, and the knowledge of the direction of the three reference coordinate axes. These initial conditions may be obtained by solving the cruise phase navigation problem with the second stage guidance system, or by using the outputs of the first stage navigation system to transfer the initial conditions to the second stage system.

2.3.1 Stage 1 Navigation Techniques

A number of navigation techniques have been considered including air data instruments, doppler radar and astrocompass, inertial systems, aided inertial systems, ground based radio aids, and navigation satellite systems. An on-board dead reckoning navigation system is necessary in all cases.

(1) True Airspeed and Magnetic Compass

One of the simplest dead reckoning navigation sensor combinations is a magnetic compass and true airspeed meter. Winds are estimated from weather data or by using a series of position fixes obtained with the available navigation aids. In the current application requiring world wide coverage, the navigation aids would be the Omega system or a navigation satellite system. The accuracy of true air-speed determined from measurements of dynamic pressure and static pressure at hypersonic speeds has not been determined. However, based upon subsonic experience this type of system is estimated to have a dead reckoning accuracy of 3 to 5% of the distance traveled. If this system were used for return navigation of the first stage (using an accurate second stage navigation and guidance system for the outbound phase) the position error after 3704 km (2000 nautical miles) would be 110to 185 km (60 to 100 nautical

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miles). Landing approach aids for the terminal 280 km (150 miles) would be required such as Tacan, VOR or an airfield located surveillance radar or tracking radar. Fuel reserves are required to compensate for the landing approach errors. The complexity of the required landing aids takes this system out of the simple category. Detailed work is not planned at this point to further define the characteristics of this system since it is generally considered obsolete as a navigation system. However, an airspeed meter is required to aid in piloting on takeoff and landing. Airspeed data, pressure altitude, and air density can also be important inputs to the guidance function to minimize performance penalties associated with avoiding flight profile constraints.

(2) Mapping Radar Position Fixes

Military aircraft of the bomber or intercontinental transport type have used airborne mapping radar to obtain position fixes and wind measurements. Accuracy after the position fix has been about 3% of the distance traveled. A mapping radar is not applicable to the current launch vehicle because of the worldwide launch requirement. The airbreathing stage flight path could be entirely over water.

(3) Doppler Radar and Magnetic Compass

Velocity information obtained from a doppler radar has been in use for some time on military and commercial aircraft. In this system, the doppler shift obtained from each of three or four ground pointing beams is used to compute the vehicle velocity magnitude and direction with respect to the vehicle primary axes. An external heading reference is used to resolve the velocity into North and East components. The accuracy of the longitudinal velocity measurement is about 0.1% for the higher accuracy systems. The accuracy of the position computation is highly dependent on the accuracy of the heading reference, which incidentally, must be aligned to the antenna beam axes to a high degree of accuracy. The accuracy of a magnetic compass can rarely be counted on to be better than 0.5 degrees, resulting in a 0.8% drift velocity error. If better accuracy is required, a high accuracy gyro heading reference is used. Tests of a commercial doppler navigation system (10) show an error of less than 2% over 90% of the flights using a compass reference. The expected error of a current military doppler navigator is 0.43% when used with a gyro heading reference and 1.7% when used with a compass heading reference. Accuracy improvement requires a better heading sensor and improvement in the level attitude reference.

(4) Doppler Radar and Astrocompass

Improved heading data can be obtained with an astrocompass used to track stars at night and the sun or moon during the day. Limitation to night star tracking simplifies the star acquisition problem when a moderate accuracy level attitude reference is used. Azimuth accuracy is a direct function of level accuracy for star elevation angles above the horizon; there is a one-to-one correspondence at a 45° elevation angle. In a supersonic aircraft installation a bubble tracker window would not be allowable and the feasible elevation angles with a flush window installation would be 45° or greater. Typical astrocompass heading accuracies are 0.1° with relatively low cost and performance level references. An astrocompass-

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doppler radar navigation system has an estimated accuracy of 0.2 to 1%. The accuracy of the computer for the dead reckoning integration of velocity components becomes a factor. Digital integration becomes desirable to achieve the potential available in the sensors.

Doppler radar and astrocompass sensors are limited in accuracy by the level reference. To maintain level during aircraft accelerations and turbulent motions requires acceleration measurement to torque the platform. Schuler tuning of the platform leveling servos makes the platform insensitive to accelerations. Schuler tuning is obtained by double integration of the acceleration measurements and adjustment of the platform torquing scale factor so that the angle that the platform is turned is equal to the earth central angle corresponding to the distance traveled. This maintains the platform at the local level reference condition. Except for the accuracy of gyro and accelerometer components, the level platform that is required to improve Doppler-astrocompass accuracy has the functional capability of a pure inertial navigator. Improvements in gyro and accelerometer state-of-the-art have made inertial navigators competitive in cost and complexity and potentially better in accuracy. The trade between a Doppler radar - astrocompass system and a pure inertial system has been resolved in favor of the pure inertial system for a number of recent applications.

(5) Inertial Navigators

The high velocity and relatively short mission time of the cruise launch vehicle make the choice of an inertial navigation system attractive. The errors in a cruise inertial navigation system are predominately time dependent and have relatively little dependence on the distance travelled. The position errors in an undamped inertial navigation system tend to grow without bound. Table 2.3-1 shows the single axis errors vs time for three accuracy classes of inertial navigation systems. Only predominate errors have been included.

(6) Aided Inertial Systems

Because the position error due to gyro drift increases in an unbounded fashion in an inertial navigation system, auxiliary sensors are used to damp the system error and/or reset the vehicle position. Means employed include (1) doppler damping, (2) stellar damping, and (3) position fixes. In the absence of position fixes, doppler and stellar sensors damp the system error output, but do not eliminate it completely. Position fixes are used to update position and also damp the inertial navigator error, if desired. Table 2.3-2 from the Boeing C5-A guidance study (11), shows the errors for four modes of operation of a stellar-inertial-doppler system. For the quality of inertial system chosen (which is a high accuracy system), the doppler information improves the system by only 33%.

The combination of an inertial navigator and a Doppler radar takes a different detailed form depending on the relative accuracy of the two devices under consideration. The inertial system has low frequency noise due to gyro and accelerometer errors. The fundamental period of the error effects is 84 minutes. The Doppler radar has high frequency noise due to variations in terrain reflections and relatively small amount of low frequency noise. The combination of the two devices in a velocity damping mode of operation is a filter design problem to reject as much noise as possible.

An inertial quality platform is required to make a daylight star tracker feasible. A very narrow star tracker field of view is required to reject background scattered light so that the star signal is greater than the background. The initial angular error must be small so that the star search and acquisition problem can be solved in a reasonable time. Thus, the star tracker becomes an auxiliary device for correcting gyro drift and the inertial navigator is the basic functional component. This is the evolutionary form of the night-time astrocompass.

In a stellar-inertial system, a star tracker is used to measure the angles to two stars to provide a system correction. Two methods can be used to apply the correction. One method is to assume the platform level reference is perfect and correct the position, and the other method is to assume the position is perfect and correct the platform. Operational considerations usually dictate which method to use. For example, if the stellar tracker is used to align the platform immediately after takeoff, the second method is used, since the position error will be small compared to the alignment error. The system error response for both methods is the same. Figure 2.3-1 (12) shows the single axis error block diagrams for the stellar monitored inertial navigation system. Figures 2.3-2 and 2.3-3 show the single channel error response due to gyro random drift and stellar tracker random error. It should be noted that this analysis assumes continuous tracking. When only one tracker is used, this is impossible, since two stars are needed for a stellar update. If, however, the frequency of stellar updating is high compared to the system error response time, the approximation to continuous tracking is valid.

Aided Inertial vs. Pure Inertial

For the short mission time of the first stage, it appears that a moderate-to-high accuracy inertial navigation system is adequate. Such a system would have an error of 1.85 km per hour (1 N.M. per hour) or less. The addition of doppler or stellar information would be based mostly on operational and reliability considerations.

Doppler damping does not look attractive for the following reasons:

(1) Current operational doppler systems (e.g., B-58) have an altitude capacity of 21 km (70,000 ft.) and speeds to 2800 km per hour (1500 knots). This does not meet the first stage requirements; thus, a development program would be required which increases cost relative to other techniques with competitive accuracy.

(2) Doppler operation is not reliable over a calm sea.

(3) The improvement in accuracy over a high accuracy inertial system is marginal in this application.

The stellar-inertial system offers higher accuracy than the doppler inertial system. It also has an important operational advantage of providing an in-flight fine platform alignment after takeoff. The stellar tracker, acting as a monitor, also insures that the inertial platform is kept in a "tuned-up" condition, realizing the full capability of the inertial system. Disadvantages in using a stellar tracker include the higher cost, increased computer capacity required, and problems in providing optical windows in the launch vehicle. Shock wave diffraction errors and errors due to thermal gradients in the window may also be critical problems for a Stage 1 stellar-inertial system.

The addition of a doppler radar or a star tracker aid will have a greater influence for the 9,260 km (5000 nautical miles) cruise mission.

(7) Advanced Inertial Systems

One of the most promising of the advanced inertial system concepts is the electro-statically suspended gyro (ESG) system, and in particular, the strapdown ESG system. The accuracy of a strapdown ESG system is comparable to that of contemporary platform systems. Table 2.3-3 shows a comparison between a conventional gyro platform system using conventional gyros and an ESG strapdown system using currently available technology (13).

Error Correction

A single position fix will correct the accumulated effect of inertial navigator error sources on position. If the initial position is also known (the take off coordinates for example) then the position error at the fix point allows an estimate of the average velocity error when the elapsed time is observed. Additional position fixes provide the opportunity to obtain additional estimates of error sources. Recursive data processing techniques (the Kalman filter) have been applied for parameter estimation. An alternate digital filtering technique, conventional Bayes weighted least squares parameter estimating, has been shown to be equivalent, e.g., reference 2. The choice of digital filter technique is largely a matter of data processing convenience.

The digital filter approach can also be used for velocity mixing, star tracker data processing, or some other sensor application. The digital filter makes best use of measurements made at different times along the path and takes into account cross-coupling effects in the propagation of errors along the path. This model can suggest sensor types and places along the path to make measurements for error components of interest.

Digital filters also smooth noisy redundant data. This process can be thought of as curve fitting to noisy data samples. The effect of bias errors and changing errors must be included to avoid unrealistic conclusions that can be obtained by analyzing the smoothing of random errors only.

(8) Radio Navigation

OMEGA (References 3-5)

The OMEGA system is the only ground radio navigation system that can provide near world-wide coverage. The OMEGA system is a hyperbolic radio navigation system which uses a very low frequency (VLF) carrier of 10.2 kilo-hertz. The low propogation attenuation of VLF waves make possible global coverage using only eight stations. The very long base-lines of 9000 to 11,000 km (5000 to 6000 nautical miles) results in available line-of-position crossing angles of not less than 60 degrees, which allows accurate fixes anywhere on the Earth.

The OMEGA signals consist of 10.2 kilo-hertz continuous wave pulses transmitted sequentially from each of eight stations. All stations are synchronized. The pulse length and its position in the 10-second period identifies the station in the navigator's receiver. The entire system is synchronized on UT-2 time with the 10 second period beginning at 0000 hours and repeating at 10-second intervals.

The RMS fix accuracy obtainable from an 8-station global network is 0.9 km (0.5 miles) daytime and 2.2 km (1.2 miles) night time. The present experimental system, with stations in Hawaii, Canal Zone, New York, and England, has accuracies which approach these values in portions of the coverage area. The limiting factor in OMEGA system accuracy is the stability of the signal propagating medium. During transition hours near sunset and sunrise, the phase of the VLF signal changes rapidly, resulting in high instantaneous errors during this period if accurate corrections are not known.

Application of OMEGA to the Mission

It is not expected that the OMEGA system will be applicable as a prime navigation mode forthe following reasons:

1. Since at least 10 seconds are required for a fix, during which a Mach 7 vehicle travels 24 km (13 N.M.), a dead reckoning system must be provided.

2. System fix accuracy during twilight and dawn period is not adequate.

3. For the relatively short cruise vehicle missions, the accuracy of an OMEGA fix is no better than the accuracy of a good inertial dead reckoning system.

(9) Navigational Satellites (References 6-9)

Of the navigation satellite concepts studies so far, only one has reached the testing phase, this being the Navy navigational satellite system. The Navy system uses the measurement of the doppler shift of 200 mega-hertz carriers transmitted from two pairs of orbital positions. The accuracy of the fix depends on the tracking time, but approaches 0.5 km (0.25 NM) for airplane tracks. The accuracy of the fix is highly dependent on the knowledge of vehicle velocity. This velocity information would most likely come from a dead reckoning system such as an inertial navigator or a doppler radar system. Also, the dead reckoning system must continuously update vehicle position during the tracking period. Three satellites are now deployed in 1100 km (500 N.M.) circular polar orbits. A synopsis of current navigation satellite concepts is shown in Table 2.3-4. The utility of using navigation satellite position fixing is somewhat limited for a hypersonic cruise vehicle because of the relatively low fix frequency and the requirement for a dead reckoning system to augment the fix. Since the navigation satellite fix is dependent on the dead reckoning system, it cannot be used as a backup to it. When the accuracy of the dead reckoning system cannot meet mission requirements, the navigation satellite fix begins to look more attractive.

The OMEGA system and navigational satellites will be considered for updating low accuracy onboard navigation systems and for reliability improvement and checking of high accuracy navigation systems.

(10) Short Range Navigation

For navigation during the takeoff and climb, and descent and landing phases of the flight, use will be made of ground navigational aids such as VOR, TACAN, ground radar tracking, and ILS (instrument landing system) systems. In addition, use is often made of weather radar for position fixing.

(11) Altimeter

For all of the navigation systems considered, an altimeter is necessary for the Stage 1 vehicle to prevent the buildup of large altitude errors or to avoid excess system complexity. Both pressure altitude, obtained from the air data system, and true altitude, obtained from a radar altimeter should be provided.

A large number of alternate Stage 1 navigation concepts have been briefly described. A number of these concepts can be eliminated from the considerations given in this section. The next section considers the Stage 2 azimuth alignment problem and 2.3.3 then develops comparative error analyses for injection into a rendezvous transfer orbit. Payload penalties due to error, reliability, and cost data are analyzed and then combined in tradeoffs in Section 2.8. These tradeoffs are the basis for selecting the most promising navigation concept. Anticipating the result, pure inertial navigators have adequate accuracy and minimize launch vehicle installation problems. Navigation satellites and the Omega system are of value for updating and inflight monitoring of an advanced navigator.

TABLE 2.3-1 COMPARISON OF SINGLE AXIS ERRORS USING LOW, MEDIUM, AND HIGH ACCURACY INERTIAL NAVIGATION SYSTEMS 2.13 KM/SEC (7000 FT/SEC) VEHICLE VELOCITY

	Error After 21 Min		Error After 42 Min		Error After 63 Min	
	KM	(N.M.)	KM	(N.M.)	KM	(N.M.)
Low Accuracy						
Accel. Bias = 10^{-4}g	.64	(.344)	1.28	(.688)	.64	(.344)
Accel. Scale Factor = .1%	1.78	(.963)	0	0	1.78	(.963)
Gyro Drift = .1 deg/Hr.	1.41	(.76)	7.78	(4.2)	14.1	(7.6)
RSS	2.35	(1.27)	7.87	(4.25)	14.2	(7.66)
Medium Accuracy						
Accel. Bias = $5(10)^{-5}$g	.32	(.172)	.64	(.344)	.32	(.172)
Accel Scale Factor = .05%	.89	(.481)	0	0	.89	(.481)
Gyro Drift = .01 deg/Hr	.14	(.076)	.78	(.42)	.14	(.76)
RSS	.96	(.517)	1.00	(.542)	1.70	(.915)
High Accuracy						
Accel. Bias = 10^{-5}g	.063	(.034)	.126	(.068)	.063	(.034)
Accel. Scale Factor = .01%	.178	(.096)	0	0	.178	(.096)
Gyro Drift - .001 deg/Hr	.015	(.008)	.078	(.042)	.141	(.076)
RSS	.19	(.102)	.15	(.08)	.235	(.127)

TABLE 2.3-2 MODE ACCURACIES - STELLAR INERTIAL DOPPLER SYSTEM

MODE	POSITION ACCURACY (CEP)
Stellar-Inertial-Doppler	0.91 km (3000 Feet)
Doppler-Inertial	0.93 km/hr (0.5 NM/HR) Max. 1st Hr. 2.8 km/hr (1.5 NM/HR) Max. 13 hours Thereafter
Stellar-Inertial	1.4 km (4500 Feet) After 14 hours
Inertial	1.4 km/hr (0.75 NM/HR) Max. 1st Hr. 3.7 km/hr 2 NM/HR Max. 13 Hours Thereafter
Doppler Dead Reckoning	1% of Distance Traveled
Dead Reckoning	Variable

PRIMARY ERRORS

1. Gyro Drift	.005 deg/Hr. random, level axis
2. Accelerometer Null Stability	2×10^{-5} g
3. Accelerometer Scale Factor	10^{-5} g
4. Tracker Accuracy	10 sec.
5. Doppler Ground Speed	.11%

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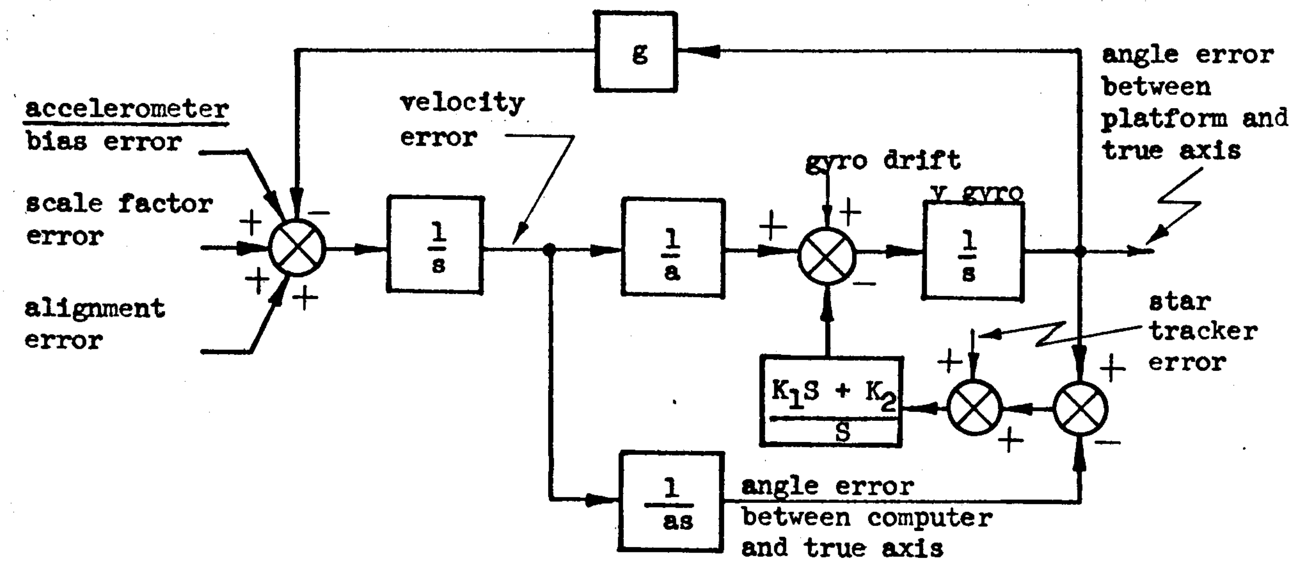
Single channel error diagram showing stellar corrections applied to the platform and gyro bias.



Single channel error diagram showing stellar corrections applied to computed position and gyro bias.

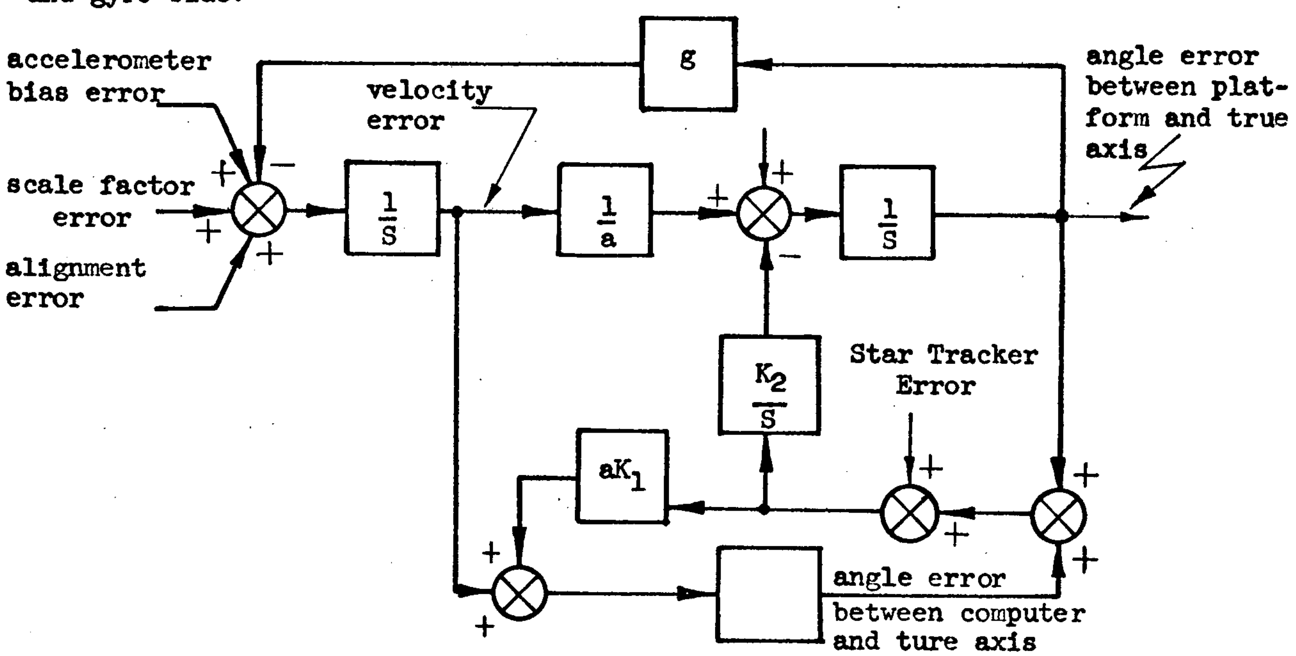


Fig. 2.3-1 Error Block Diagrams for Stellar-Inertial Navigation System

Fig. 2.3-2

Stellar-inertial position
Error vs Operating Time.
Error due to .01 deg/hr
level gyro drift, correlation
time = 1 hr.

1. Free Inertial
2. ζ = .8, τ = 1 Min.
3. ζ = .8, τ = 2 Min.
4. ζ = .8, τ = 5 Min.
5. ζ = .8, τ = 10 Min.
6. ζ = .8, τ = 30 Min.

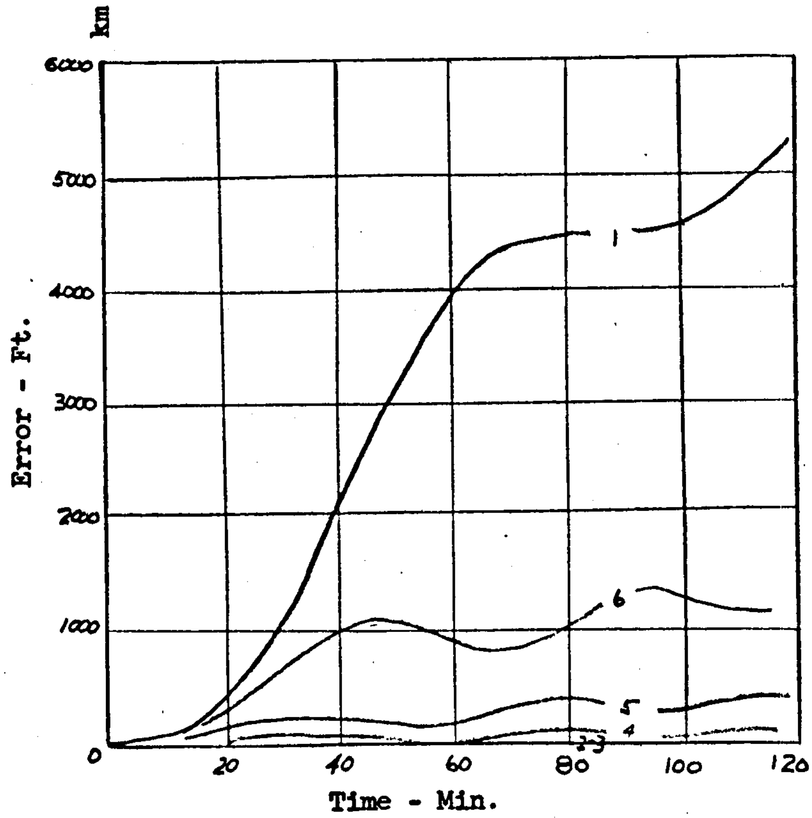


Fig. 2.3-3

Stellar inertial position
Error vs Operating Time
Error due to a 1 arc sec
tracker Error, correlation
time = 5 Min.

1. Free Inertial
2. ζ = .8, τ = 1 Min.
3. ζ = .8, τ = 2 Min.
4. ζ = .8, τ = 5 Min.
5. ζ = .8, τ = 10 Min.
6. ζ = .8, τ = 30 Min.

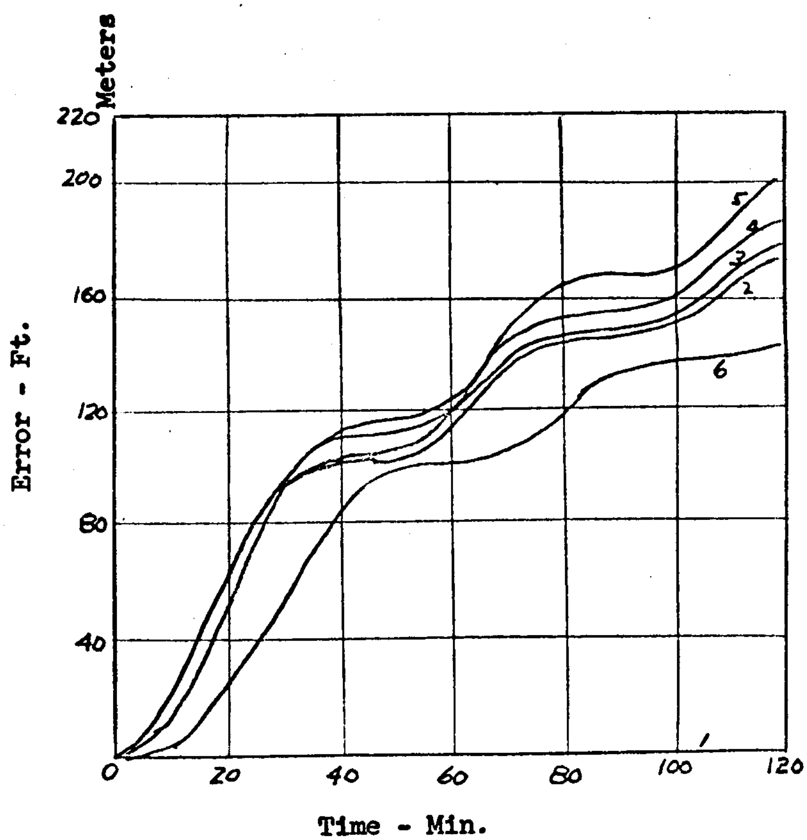


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TABLE 2.3-3 - COMPARISON OF SDOF* GYRO GIMBALED SYSTEM AND ESG** STRAPDOWN SYSTEM

UNIT/ SUB-ASSEMBLY	GIMBALED PLATFORM WEIGHT kg.	(Lbs.)	VOLUME m^3	(In^3)	POWER W	EST. MTBF HRS.	STRAPDOWN ELECTROSTATIC GYRO SYSTEM WEIGHT kg.	(Lbs.)	VOLUME m^3	(In^3)	POWER W	EST. MTBF HRS.
Inertial Platform	16	(35)	.015	(900)	46	1375	-	-	-	-	-	-
Sensor Package	-	-	-	-	-	-	6	(14)	.0033	(210)	15	22,000
Gyro & Accel. Electronics	5	(12)	.005	(305)	28	6400	4	(9)	.0054	(330)	43	4,100
Power Supply & Gimbal Electronics	16	(36)	.013	(815)	96	3830	-	-	-	-	-	-
Power Supply	-	-	-	-	-	-	7	(15)	.0056	(340)	18	7,800
Computer	14	(32)	.016	(950)	84	4300	20	(45)	.022	(1340)	118	3,200
SYSTEM TOTALS	51	(114)	.0487	(2,970)	254	725	38	(83)	.0364	(2,220)	294	1,370

* Single-degree-of-freedom
** Electrostatic Gyro

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TABLE 2.3-4 NAVIGATION SATELLITE CONCEPT COMPARISONS

SYSTEM	MEASURING TECHNIQUES	COVERAGE/PRECISIONS
Doppler NavSat (Limited System of 3 Satellites Now Deployed)	Doppler shift of 200 mega-hertz transmitted from two pairs of orbital positions (10 min. track) defines two intersecting hyperbolas for position fix. Active mode (user transmits for control center computation) not inherent, but could be attached.	Requirements for world-wide coverage not determined. Experiments show precision as good as ± 46 km (± 25 n.m.) for air craft tracks. Long tracks (as taken by Polaris sub, for example) may fix position within tens of meters.
Radar-Ranging Navsat (study)	Travel times of pulse signals from two satellites to user determine slant range and trilateration gives position fix. A/C must supply altitude or third satellite must be used. Active and passive modes.	24 satellites in 10,400 km (5600-n.m.) orbits and 6 control centers for world-wide continuous coverage. Precision probably better than ± 1.8 km (± 1 n.m.) except directly under satellite.
Interferometer Navsat (study)	Incidence angle of signal from user is measured by two pairs of 30 meter (100-ft) interferometers on satellite to fix position. A/C altitude determined by range pulse sent to satellite from A/C. Active mode only.	8 satellites in 11,000 km (6000-n.m.) orbits, 6 control centers, 18 stations for global, continuous coverage. ± 0.05 milliradian attitude measurement for ± 1.8 km (± 1 n.m.) precision.
Swept-Beam Navsat (Study)	Two orthogonal (X and Y) beams swept across earth by spinning satellite are timed between cross of reference point and cross of user to give position fix. Basically passive, but active mode possible.	Satellites and control centers undisclosed. For spin rate of 180 rpm, ± 0.025 milliradian measurement for ± 1.3 km (± 0.7 N.M.) precision. Medium and synchronous altitudes.
Phase Differencing And Ranging Navsat (study)	Relative phase of signals received by user from two satellites defines two intersecting circles to give position fix. Active mode. Two passive modes possible; 1) with stable oscillator at user, circular fix can be computed, or 2) difference of relative phases from two pairs of satellites (3 navsats total) can give hyperbolic fix as with Omega.	8-10 satellites at synchronous altitudes plus several control centers required for broad worldwide coverage. Simpler ground-based local centers and repeaters for extending usage. ± 150 m (± 500 ft) precision possible for absolute fix. (± 15 m, ± 50 ft. in "relative" fix mode).

2.3.2 Stage 2 Alignment Methods

The specification of initial conditions during preflight can be accomplished for a fixed base to almost any required accuracy. The reference coordinate frame orientation can be established within 5 to 10 arc seconds using conventional optical techniques. Initial position and velocity are determined by the fixed base. Launch from a moving base greatly increases the complexity of establishing the initial conditions although this is now done routinely with such systems as Polaris and aircraft operating from carriers.

Leveling

In-flight leveling of a stage 2 inertial platform can be accomplished by using a stage 1 master platform as a reference. The difference in velocity between the master platform and secondary platform is used as an input to the torquers of the secondary platform gyros. The secondary platform is torqued until the two platforms have the same velocity output when averaged over the time constant of the leveling loop. In a third order mechanization gyro, drift error effects are eliminated and the accuracy of level is determined by the secondary platform accelerometer bias. A 10^{-4}g bias gives a 20 second of arc leveling error. Structural oscillations between the two platforms and master platform noise errors may double the error. Thus, a level accuracy of 40 seconds of arc can be expected at separation of stage 2 with an inexpensive stage 2 platform.

Azimuth Alignment or Transfer

The critical technical problem with a master platform - secondary platform configuration is azimuth alignment of the secondary platform. A number of alternate transfer methods and alignment methods can be considered.

(1) Ground-based alignment
Pre-takeoff leveling and azimuth alignment of the stage 2 inertial platform requires a low drift rate azimuth gyro. Level of stage 2 can be held relative to a stage 1 master platform by the leveling mode. After a 40 minute flight to the staging point the azimuth accuracy for various gyro drift rates is:

Gyro drift rate (degrees/hour) *	Azimuth error degrees
1.	0.7
0.1	0.07
0.01	0.007
0.001	0.003

(2) Stage 2 star tracker
The use of a star tracker for azimuth measurement requires a precision window and additional computer capacity. A roll maneuver of stage 2 may be required to put the desired star in the star tracker search field of view.

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Star acquisition 20 seconds after beginning search is expected for a good design. The designated star is chosen with care to avoid acquisition of the wrong star. Use of a star tracker for a manned payload would be particularly valuable to check the operation of gyros and to provide a backup attitude reference. 10 seconds of arc angle measuring accuracy can be obtained.

(3) Synchro azimuth transfer
Synchro transfer of vehicle azimuth angle from a master platform to a secondary platform is inaccurate because of structural deflections. Several degree deflections are typical in turbulent air. Structural weight minimization requirements make accurate synchro azimuth transfer impractical.

(4) Optical transfer
Optical azimuth transfer between two platforms requires an unobstructed line of sight. Accomplishing this with optical instruments or mirrors on the inner gimbal of a platform is impractical for an arbitrary launch vehicle heading. Synchro transfer can be corrected with an optical measurement of the structural deflection between the structural bases on which the platforms are mounted.

(5) Gyro-compassing
In flight gyro-compassing with an accurate azimuth gyro on a stage 2 platform is limited by the accuracy of vehicle velocity. Better performance is obtained with preflight alignment with the accurate azimuth gyro required for gyro-compassing and the relatively short flight time of the basic mission. Gyro-compassing is one of the potential modes of preflight azimuth alignment.

(6) Velocity change matching
Azimuth transfer can be accomplished by comparing the velocity change measured by the master platform with the velocity change measured by the secondary platform and attributing differences to azimuth errors. A velocity change or vehicle turn is required for the comparison. Three error sources determine the resulting azimuth accuracy. (a) the master platform azimuth error, (b) vehicle azimuth oscillations (say 3 m/sec, 10 fps) divided by the input velocity change (say 0.9 km/sec, 3000 fps) gives an azimuth error in radius (3×10^{-3} radians = 0.2° in the example), (c) the ability of the secondary platform to hold the alignment after the maneuver.

(7) Position data use for azimuth alignment
Position data obtained from the master platform is compared with the position data output of a stage 2 inertial navigator using a weighted least squares digital filter (or Kalman filter). The filtering process gives estimates of stage 2 navigator errors including azimuth alignment errors. Gyro bias drift errors can also be estimated. For a single turning maneuver comparison the position data method reduces to the velocity change matching result. However, the position data digital filter method has several advantages. Data over a longer time interval is used so that smaller maneuvers are required for a given azimuth error; this may provide

a significant operational advantage for some launch vehicle missions. Also, the effect of vehicle azimuth oscillations can be estimated using multiple position data comparisons over several oscillations. This potentially could improve the azimuth accuracy compared to velocity match methods. Further detailed study is required to establish the magnitude of the accuracy advantage.

The selection between Stage 2 azimuth alignment techniques can be made from a study of the tradeoffs in Section 2.8. It is hown there that accurate preflight alignment of a Stage 2 inertial navigator and use of high quality gyros to maintain the alignment is a cost-effective answer.

2.3.3 Stage 2 Navigation Systems and Error Analysis

A number of alternate Stage 2 navigation and guidance configurations are defined to obtain a range of accuracy performance. Alternate alignment techniques are used: (1) alignment transfer from a Stage 1 master platform, (2) preflight alignment of the Stage 2 inertial platform, and (3) alignment with a Stage 2 star tracker. An error analysis has been made to describe the errors at the point of injection into the transfer orbit at 74 km (40 NM) altitude. The Stage 2 velocity gained during the transfer injection thrust period is approximately 6 km/sec (19,500 FPS.) The accuracy performance of each configuration is summarized by the one sigma vector velocity error at the transfer orbit injection point. Tables 2.3-5 and 2.3-6 list the specifications for the set of systems considered. They are listed in the order of increasing boost cutoff velocity accuracy. The accuracy is dependent upon the method of updating and aligning the Stage 2 system before separation, as well as on the specifications of Stage 2 guidance components. It varies from 41 m/sec (135 FPS) through 0.67 m/sec (2.2 FPS) over the 13 systems, as shown in Table 2.3-7.

System 1 has a low accuracy Stage 2 system with open loop guidance. It is updated before separation by a medium accuracy 1.85 km/hr (1 NM/Hr.) Stage 1 system. The update includes changes in guidance parameters to compensate for stage time deviations. The second Stage 2 system is also of low accuracy but has closed loop guidance. It is updated by a low accuracy 18.5 km/hr (10 NM/Hr) Stage 1 system. The third Stage 2 system is the same as the second system, but is updated by 1.85 km/hr (1 NM/Hr.) Stage 1 system. Systems 4 and 5 are updated before separation by a 1.85 km/hr. (1 NM/Hr.) Stage 1 system and have progressively better gyro and accelerometer specifications. Stage 2 systems 6, 7 and 8 have still better gyro and accelerometer specifications. They are aligned before launch and not updated on the flight. Comparable accuracy is achieved in System 9 using lower quality gyros and a star tracker on Stage 2. Improvement of the gyro and accelerometer accuracy on systems 10 and 11 gives very little boost cut-off velocity accuracy improvement. However, the Stage 2 system is used for navigation out and a 18.5 km/hr (10 NM/Hr.) Stage 1 system has been selected for the rendezvous function trades. System 12 uses the same quality gyros and accelerometers as system 7, that is 0.01°/Hr. and 3×10^{-5} g. An accuracy improvement in alignment is expected, by mixing the outputs of a 0.185 km/hr (0.1 NM/Hr.) Stage 1 system and the Stage 2 system during the cruise out.

The highest accuracy Stage 2 system is system 13, using 0.001°/Hr. gyros and 3×10^{-5} g accelerometers. The Stage 2 system is used for navigation out and a 18.5 km/hr (10 NM/Hr.) Stage 1 system is postulated. An 0.67 m/sec (2.2 FPS) vector velocity error is obtained. The predominant errors are due to alignment. Thus, a very high accuracy star tracker would improve Stage 2 performance. This refinement will not be considered at this point.

System hardware weight versus performance is one of the inputs in the study of payload weight penalties. Table 2.3-8 lists the weights of Stage 2 hardware components for a representative sample of low, medium, and high performance systems. The only weight difference among these systems occurs in the computer weights, and that varies by only 7.3 kg (16 pounds) from minimum to maximum capability. The variations in complexity are assumed to be entirely in the type of equations implemented, with the computer interface requirements remaining constant. Similarly, displays and controls are assumed to be independent of accuracy performance. A qualitative argument supporting this view is that the operational and safety functions in which the pilot plays a role do not vary with system performance capability. Therefore, the displays and controls for these functions are essentially the same for all systems. The inertial platform weight is also independent of performance specification.

The accuracy data generated in this section is used as an input to the performance penalties developed in Section 2.5 and the tradeoffs of Section 2.8.

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Table 2.3-5 Stage 2 Navigation and Guidance System Alternate Specifications, Part 1

System #	Stage 1 System Specification	Stage 2 State Vector at Staging: Method	Accuracy Km	Pos. (NM)	m/sec	Vel. (FPS)	Platform Alignment at Staging: Method	Accuracy Level	Azimuth
1	(1 NM/Hr.) 1.85 km/hr	Stage 1 Transfer	1.3	(.7)	1.5	(5.)	Transfer	0.1°	0.1°
2	(10 NM/Hr.) 18.5	Stage 1 Transfer	1.3	(7.)	15.0	(50.)	Transfer	0.1°	0.1°
3	(1 NM/Hr.) 1.85	Stage 1 Transfer	1.3	(.7)	1.5	(5.)	Transfer	0.1°	0.1°
4	(1 NM/Hr.) 1.85	Stage 1 Transfer	1.3	(.7)	1.5	(5.)	Transfer	0.1°	0.1°
5	(1 NM/Hr.) 1.85	Stage 1 Transfer	1.3	(.7)	1.5	(5.)	Transfer	0.015°	0.1°
6	-	Stage 2 Inertial	1.3	(.7)	1.5	(5.)	Preflight	.01	.01
7	-	Stage 2 Inertial	1.3	(.7)	1.5	(5.)	Preflight	.01	.01
8	-	Stage 2 Inertial	1.3	(.7)	1.5	(5.)	Preflight	.008	.008
9	(1 NM/Hr.) 1.85	Stage 1 Transfer	1.3	(.7)	1.5	(5.)	Star Tracker	.006	.006
10	-	Stage 2 Inertial	.18	(.1)	0.15	(.5)	Preflight	.001	.001
11	-	Stage 2 Inertial	1.3	(.7)	1.5	(5.)	Star Tracker	.006	.006
12	(0.1 NM 0.185)	Kalman Filter Transfer	.18	(.1)	0.15	(.5)	Kalman Filter Transfer	.006	.006
13	-	Stage 2 Inertial	.18	(.1)	0.15	(.5)		.003	.003

Table 2.3-6 Stage 2 Navigation and Guidance System Alternate Specifications, Part 2

System #	Stage 2 Guidance Error	Inertial System Component Specifications			
		Gyro Drift		Accelerometer	
		Random	G-Sens	Bias	Scale Factor
1	Open Loop: 1% Propulsion System Deviations	0.1°/Hr.	1°/Hr./g	10^{-3}g	10^{-3}g/g
2	Closed Loop: Guidance Errors Negligible	1°/Hr.	1°/Hr./g	10^{-3}g	10^{-3}g/g
3		1°/Hr.	1°/Hr./g	10^{-3}g	10^{-3}g/g
4		.1°/Hr.	1°/Hr/g	10^{-4}g	10^{-3}g/g
5		.1°/Hr.	.1°/Hr./g	10^{-4}g	10^{-4}g/g
6		.02°/Hr.	.2°/Hr./g	10^{-5}g	5.10^{-4}g/g
7		.02°/Hr.	.2°/Hr./g	10^{-5}g	10^{-4}g/g
8		.01°/Hr.	.01°/Hr./g	3.10^{-5}g	3.10^{-5}g/g
9		.1°/Hr.	.1°/Hr./g	10^{-4}g	10^{-4}g/g
10		.002°/Hr.	.1°/Hr./g	10^{-5}g	10^{-4}g/g
11		.01°/Hr.	.01°/Hr./g	3.10^{-5}g	3.10^{-5}g/g
12		.01°/Hr.	.01°/Hr./g	10^{-4}g	10^{-4}g/g
13		.001°/Hr.	.01°/Hr./g	3.10^{-5}g	3.10^{-5}g/g

Table 2.3-7

Stage 2 Alternate Systems One-Sigma Cut-off Velocity Error

System #	One Sigma Velocity Error Meters/Second (Feet Per Second) Down Range		Vertical		Cross-Plane		RSS	
1	10.7	(35)	36.6	(120)	15.2	(50)	41.1	(135)
2	11.0	(36)	24.4	(80)	16.7	(55)	31.7	(104)
3	10.7	(35)	19.2	(63)	16.5	(54)	27.4	(90)
4	7.3	(24)	14.3	(47)	14.6	(48)	22.0	(72)
5	1.5	(5)	2.7	(9)	10.7	(35)	11.3	(37)
6	3.4	(11)	3.4	(11)	2.4	(8)	5.2	(17)
7	1.2	(4)	3.0	(10)	2.4	(8)	4.0	(13)
8	0.7	(2.3)	1.8	(5.8)	0.95	(3.1)	2.1	(6.9)
9	0.85	(2.8)	1.7	(5.7)	0.67	(2.2)	2.1	(6.8)
10	0.73	(2.4)	1.4	(4.6)	1.0	(3.4)	1.9	(6.3)
11	0.61	(2.0)	1.7	(5.5)	0.46	(1.5)	1.9	(6.1)
12	0.46	(1.5)	0.73	(2.4)	0.70	(2.3)	1.2	(3.8)
13	0.30	(1.0)	0.46	(1.5)	0.40	(1.3)	0.67	(2.2)

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Table 2.3-8

STAGE 2, SYSTEM WEIGHT TABLE KILOGRAMS (POUNDS)

System	Computer		Computer Interface		Displays and Controls		Inertial Platform		Total	
4	10.9	(24)	13.6	(30)	36.2	(80)	30.8	(68)	91.5	(202)
6	14.5	(32)	13.6	(30)	36.2	(80)	30.8	(68)	95.2	(210)
7	14.5	(32)	13.6	(30)	36.2	(80)	30.8	(68)	95.2	(210)
10	18.1	(40)	13.6	(30)	36.2	(80)	30.8	(68)	98.8	(218)
11	18.1	(40)	13.6	(30)	36.2	(80)	30.8	(68)	98.8	(218)

2.4 Guidance Concepts

2.4.1 Introduction

The guidance functions consist of the rules or equations that define the control variable commands to fly the vehicle on a desired trajectory in the presence of off-nominal conditions. The effect of alternate guidance concepts on the navigation and guidance system configuration is primarily on computer requirements. The choice of sensor types to be used is part of the navigation function considered in the preceding section, 2.3.

The objective of the first stage guidance system is to provide capability such that the utility of the cruise vehicle is maximized. This implies both functional flexibility in the guidance system and minimum demands on vehicle design for the accommodation of guidance equipment. The ideal guidance design is one in which changes in vehicle characteristics do not require guidance hardware changes and in which changes in the mission flight profile do not require changes in either hardware or software. This is practically attainable with the present state of the art in real time digital control computers. The continuing efforts in the development of micro-electronics and computer technology promise cost and weight reductions as well as increased capability. Because of these computer technology advances, a dominant factor in guidance system implementation is the exploitation of computer capability to achieve optimum system performance and flexibility. To establish the penalties (if any) with obtaining flexibility trades have been developed between performance and complexity, seeking to establish the characteristics of the optimum system with cost a factor in the optimization.

The study of guidance concepts and presentation of results falls into two parts:

- Formulation of alternate combinations of flight modes and guidance laws (section 2.4) and their comparative evaluation in terms of efficiency of fuel utilization (section 2.5).

* The implementation of the guidance laws in a system of control equations and the corresponding estimate of computer requirements, (Sections 2.4-4 - 2.4-6).

2.4.2 General Guidance Concepts

The function of guidance is to produce flight path control commands in the presence of off-nominal conditions so that desired rendezvous end conditions are met without violating the structural, heating, pilot safety, and range safety limits. The guidance function is usually implemented by flying a conservative nominal flight path through the regions dominated by the constraints in aerodynamic flight and adjusting the profile to meet end conditions in the regions where the constraints are relaxed. In the regions dominated by aerodynamic constraints the nominal flight profile, if flown with no deviations, is designed to maximize the payload in orbit while meeting the constraints.

The main differences between guidance methods lie in the techniques of solving for the control profile in the regions of relaxed constraints. The solution may be based on a simple deterministic set of equations defining the control variable commands at the present time as functions of a predicted end condition state vector. An implicit guidance law determines the control variables by a power series in the state vector deviations from nominal. An explicit law is a system of control equations that are essentially independent of the nominal. The case for implicit versus explicit forms is a question of implementation. A combination of explicit and implicit forms is suggested for the subject problem.

A simplified navigation and guidance functional diagram is given in Figure 2.4-1. The relations between navigation sensors, guidance computations, and vehicle flight path control are indicated.

A generalized functional block diagram for the guidance computation functions is given by Figure 2.4-2. Two modes of flight path control are used: (1) a nominal path control mode, and (2) an adaptive control mode. For each phase of the mission, one of these control modes is selected by mode control logic. It is based on inputs of vehicle present position and velocity from the navigation function and predetermined control limits.

The nominal path control modes are based on knowledge of the flight constraints, an environmental model, and nominal values of the launch vehicle characteristics. The vehicle attitude control commands and propulsion control commands are generated so that the vehicle attempts to follow the nominal path.

Adaptive path control modes use the current values of position, velocity and time from the navigation system to predict the end conditions $X_F(t_n)$ that will be obtained at time t_n if a control variable program $Y(t)$ is followed. These end conditions are modified and the control variables are redetermined so that the modified end conditions are attained. The adaptive modes involve processes of prediction and optimization in different mathematical forms for the flight phases where it is used. The common objective of the adaptive modes is to improve the payload efficiency compared to that obtained with nominal control in the presence of off-nominal conditions.

Guidance system functions for the 5000 n.m. cruise mission and the space rescue rendezvous mission were formulated as representative of the missions to be flown. The analysis of fuel penalties associated with specific guidance laws was conducted independently of that for navigation errors. Computer requirements, memory size, and computation speed, are also reasonably independent of navigation techniques. However, variations can occur in total computer requirements for different concepts. Therefore, an overall computer requirements study was performed. Factors which enter this study include navigation system techniques, pilot control and display functions, the interface between the first and second stage system, and the impact of techniques for achieving higher reliability through redundant modes of operation with hardware redundancy at the module rather than the system level.

The concepts which were investigated are defined in this section. The results of performance trades are presented in Section 2.5 and the computer requirements data are presented in Section 2.4.6.

<u>Figure 2.4-1</u> <u>Navigation & Guidance System Functional Diagram</u>

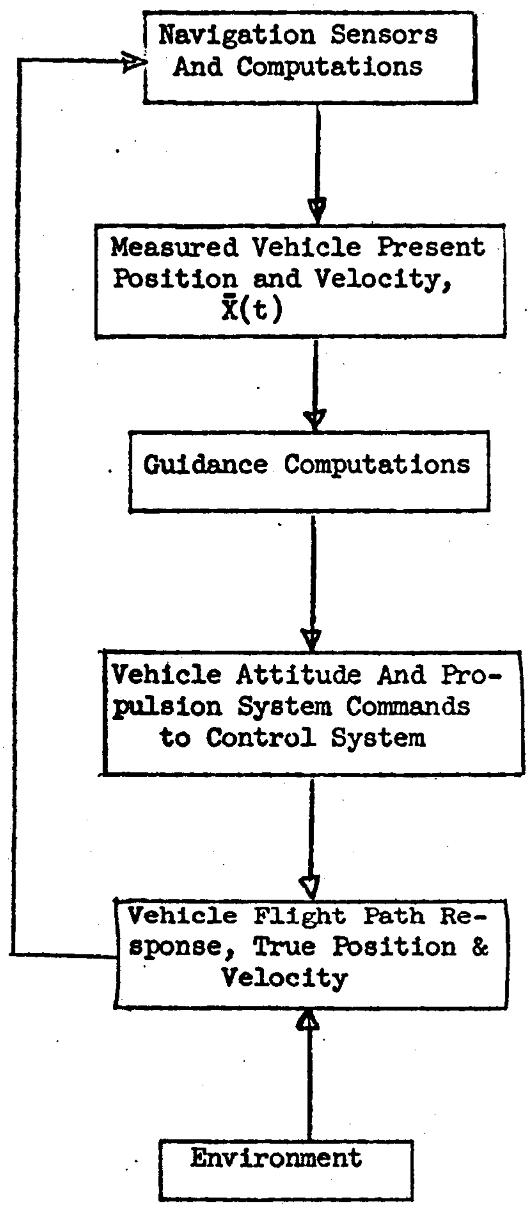

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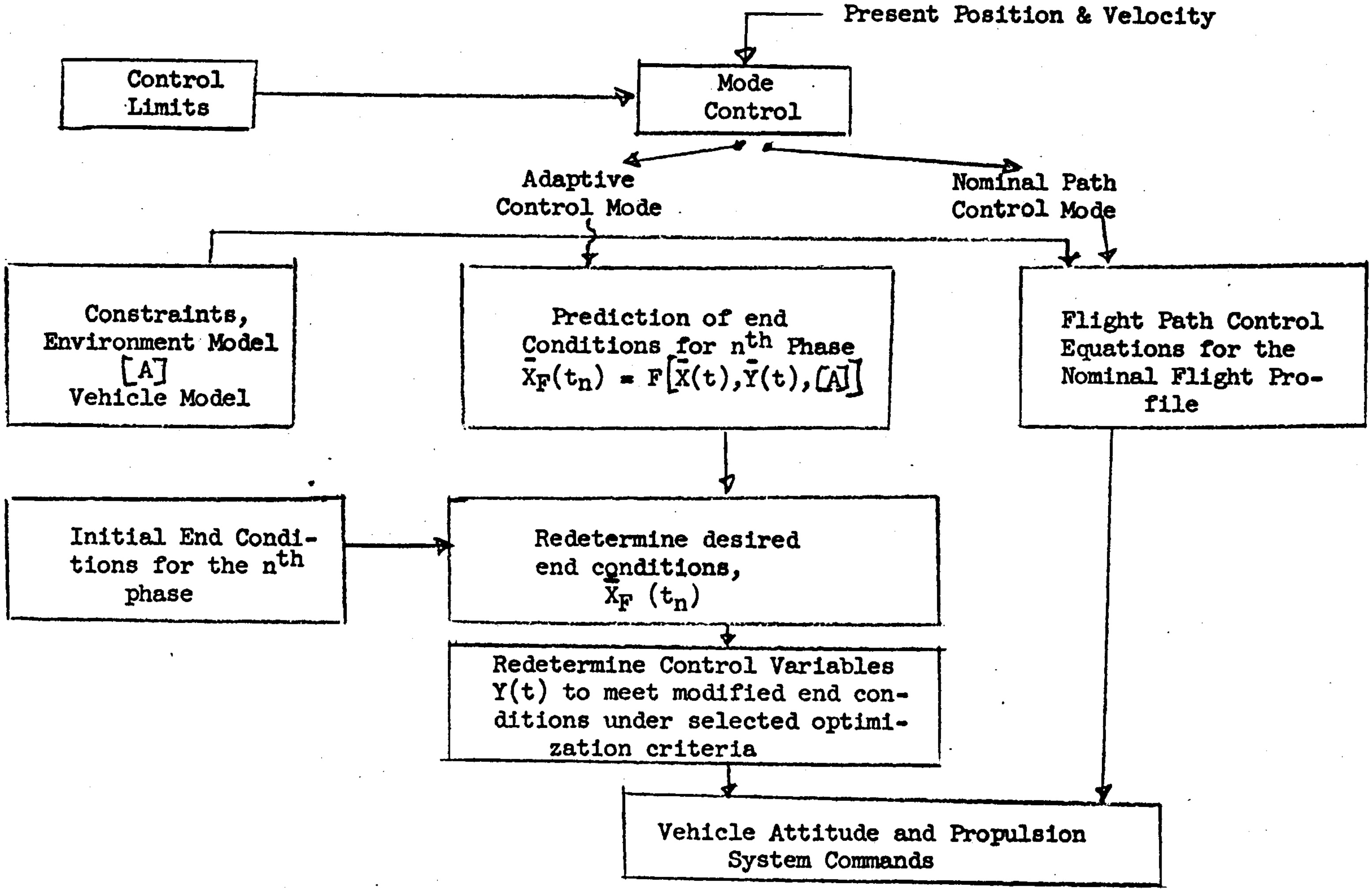
Figure 2.4-2 Guidance Computation Functional Diagram



2.4.3 Target-Vehicle Position Phasing for Rendezvous

The central problem of guidance for the rendezvous mission is the adjustment of target-vehicle relative position at injection into the rendezvous transfer orbit to compensate for unpredictable time variations on the cruise and boost flight profiles. Two solutions of the problem were studied: the parking orbit method and the direct ascent method.

The minimum energy solution of the problem involves a parking orbit coast period in an orbit below the target altitude with injection into a Hohmann transfer orbit at the exact time required to meet rendezvous conditions. The base departure time is delayed sufficiently to guarantee a lag time at staging. The rendezvous time is increased by approximately 25 minutes per minute of lag time compensated in a 300 km (161 n.m.) altitude parking orbit. Other guidance modes are required to correct for lead time situations. The alternate modes, activated in the event of a lead time at staging, include an adapted direct ascent and a coast orbit above the target altitude. Specified limits in lead time would govern the selection of the mode to be used and the decision to abort. The first stage guidance system may determine the mode on the basis of predicted vehicle-target relative position at staging. The mode prediction would be made throughout cruise with displays to the crew of lead time, mode selection, coast and orbital maneuver times, and predicted ΔV on each maneuver.

Figures 2.4-3 and 2.4-4 illustrate the flight profiles of the direct ascent and parking orbit methods of rendezvous.

The direct ascent method, which minimizes time to rendezvous, is practical if stage time is controlled within limits which can be absorbed on an adaptive second stage boost. Time deviations of 20 seconds or less can be absorbed on the boost profile for a nominal ΔV penalty. The penalty increases steeply with time deviation, so that other techniques must be defined to work in conjunction with direct ascent to accommodate large time errors. Three methods were analyzed, and are illustrated in Figure 2.4-5. The base departure time in each of the methods is biased with a lead time which is absorbed before staging. In the first method the time is absorbed on a loiter lead path after the vehicle turns into the target plane; in the second method the time is absorbed on a trombone shaped path maneuver while turning into the target plane; in the third method the course-to-steer is controlled throughout the cruise phase to adjust the vehicle-target relative position at staging. The cruise course adaptation method is significantly superior to the other two in compensating for a flight time deviations which occur

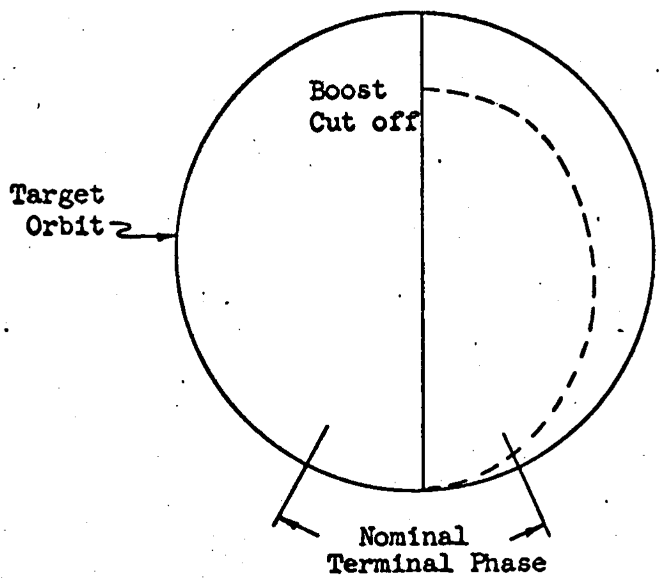


FIGURE 2.4-3 DIRECT ASCENT, ORBIT CONTROL PHASES

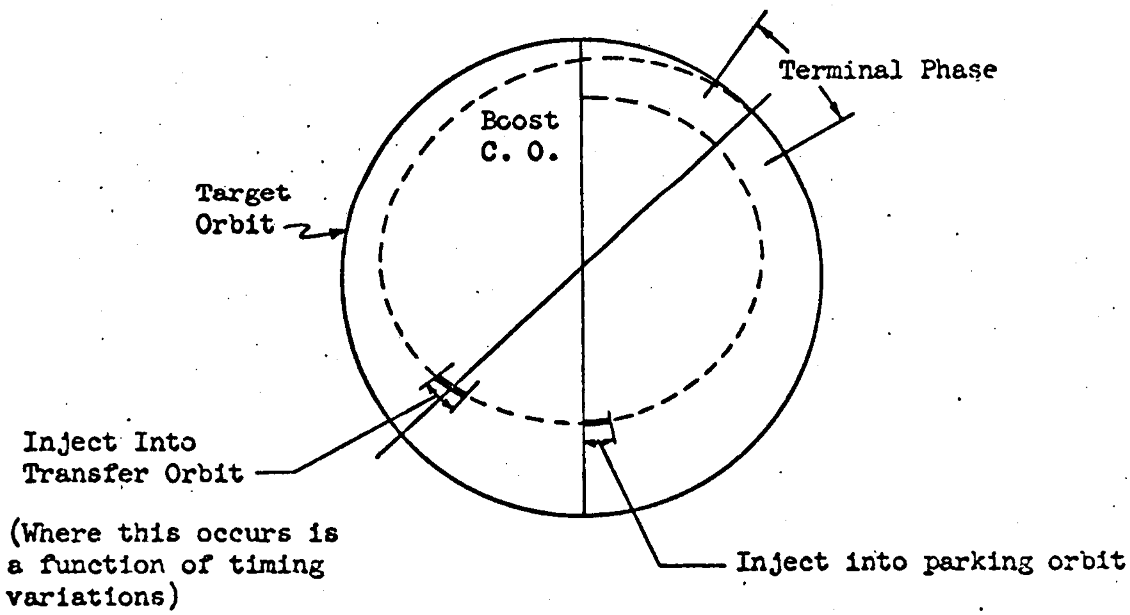


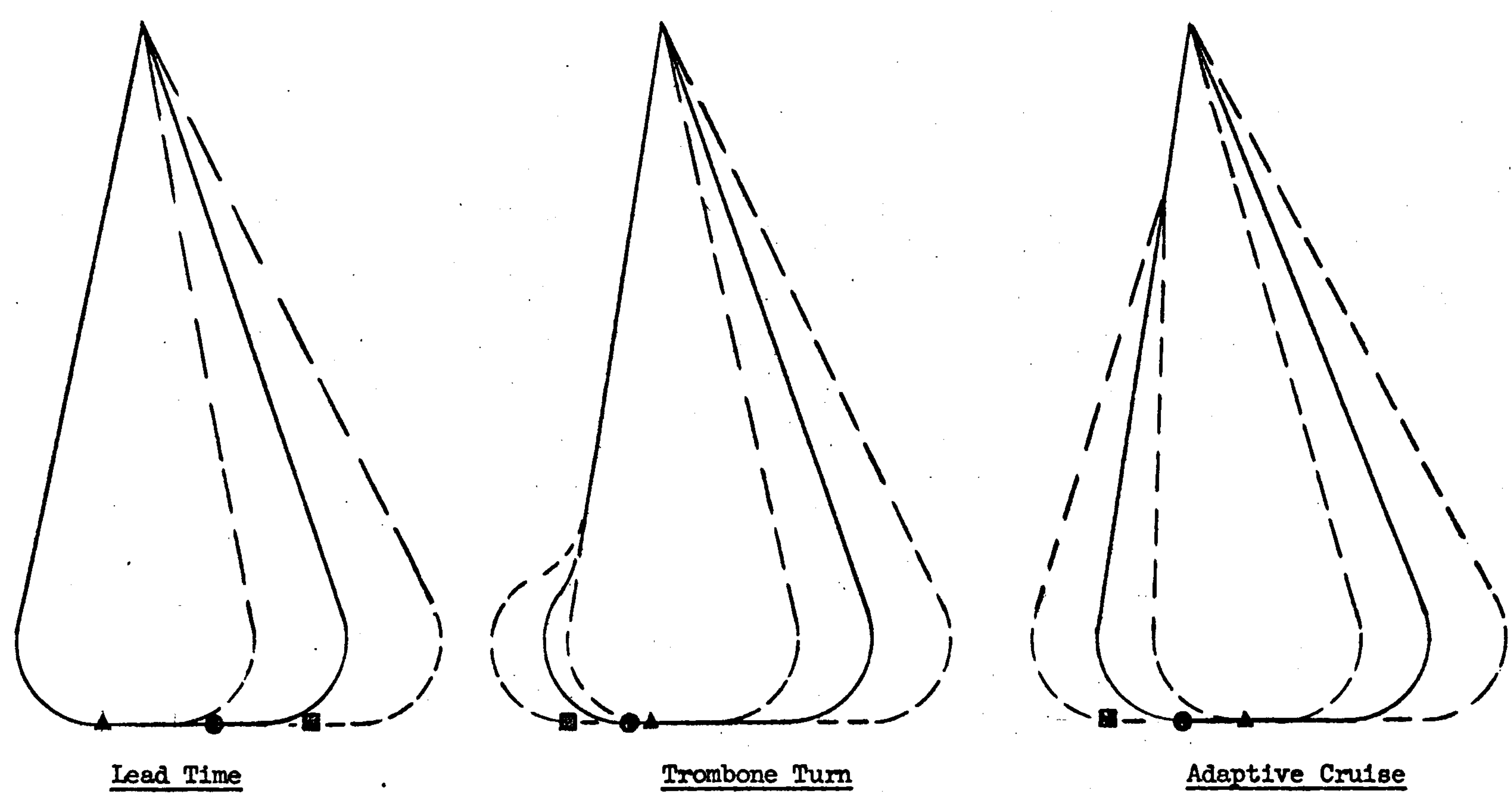
FIGURE 2.4-4 PARKING ORBIT, ORBIT CONTROL PHASES

FIGURE 2.4-5

FIRST STAGE CORRECTION METHODS FOR STAGE TIMING ERRORS

NOTES:
——— Nominal Ground Track
– – – – Off Nominal Ground Tracks

● Nominal Staging Point
■ Delayed Staging Point
▲ Accelerated Staging Point



immediately after take-off on the climb to cruise conditions. Since the largest time deviations expected occur on the climb, the method also proves to be quite effective. The base departure time is biased so that the probability of a lag time at staging is negligible within specified cruise course deviation limits. If the limits are violated, the parking orbit mode would be executed.

It will be seen that the complexity in control equations for a system which allows selection of direct ascent or parking orbit modes is not significantly greater than the complexity of each of the basic modes. The most promising approach therefore appears to be one in which the flight profile is selected on the basis of specific mission objectives or specific real time conditions. When time is critical, the flight would be directed toward minimizing time at the expense of fuel, particularly if the offset distance is less than the maximum. On routine flights, fuel minimization would generally be the controlling criterion. Since the cruise vehicle may launch the second stage at varying offset distances, a sizeable proportion of missions will require less than maximum fuel for the launch. Therefore, secondary mission objectives may be specified at the expense of first stage fuel for these missions.

2.4.4 Guidance Equations

The forms and general characteristics of the guidance equations assumed for the purposes of the computer requirements and performance studies are described in this section.

Preflight and Cruise Phases

The solution for base departure time involves basically the same computations as the solution for course to steer on the cruise phase. These two problems may therefore be integrated as different modes of the same program. The equations are in an iterative form. In the prelaunch mode an initial estimate of take off time is differentially corrected to produce the desired target-vehicle position at staging. The vehicle time on each leg of flight is predicted on the basis of nominal cruise and turn velocities. The target ephemeris is predicted using Keplerian orbital mechanics with closed form approximations for the effects of perturbations. On the outbound cruise phase, the course to steer is differentially corrected to fly the great circle path which intercepts the target plane at a desired angle at the position relative to the target required for the mode selected for rendezvous. The mode selection is a function of the predicted lead or lag time at staging. On the return leg, as on all cruise legs with a specified series of earth-fixed aim points, course-to-steer is differentially corrected to fly the great circle course between present position and the aim point.

The target orbit parameters (or the position of the base or other destinations) are the forcing parameters in the course to steer solutions. Updating target parameters or destination locations consists of simply replacing new data for old, and has no effect on the form of the equations.

A throttle control program may be used on the outbound leg of the rendezvous mission to provide a greater control range on the stage time. The throttle variation from nominal is proportional to the predicted remaining lead or lag time at staging.

Climb to Cruise Conditions

The climb to cruise conditions is governed by a predetermined pressure altitude-velocity climb profile followed by a nominal turn to the initial cruise heading determined by the preflight program. This is a flight phase in which the constraints are dominant.

The exact form of the control law is subject to continuing development.

Rendezvous Mode Prediction (Multi-Mode Capability)

Predicted position relative to the target at staging, predicted time of staging, and the corresponding lag or lead time are fundamental variables computed in the adaptive course-to-steer program. They are in turn inputs to a decision logic for selecting the rendezvous mode. The mode prediction program then determines and describes for display to the crew the event-time profile and predicted fuel requirements for all maneuvers through the target search and acquisition phase.

Prediction of events is based on the assumption that the boost profile achieves desired end conditions in the nominal time interval from the start of the pull up maneuver. Explicit Keplerian orbit equations are used to describe target and vehicle ephemerides, and an iterative routine is used to solve for the time to execute injection into the rendezvous transfer orbit in the parking orbit mode.

The mode prediction program operates periodically throughout the cruise phase, providing a continuous event profile display including the time to initiate the pull up maneuver.

Turn Into Target Plane

The turn into the target plane is initiated when the vehicle is at a given distance from the target plane, the distance being a function of the intercept angle or the total turn angle. The intercept angle is an output of the cruise course to steer program. Further study is required to determine the optimum form of the control equation, but it is expected that a nominal form of bank angle versus cross plane position and velocity will be satisfactory.

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Pull Up Maneuver

The pull up maneuver is controlled by a nominal attitude-velocity profile with staging based on specified limits in flight path angle and total velocity. The profile is highly constrained by aerodynamic pressure limits so it is expected that a profile optimization for off-nominal conditions would not result in improved efficiency. Another factor limiting the effect of optimization in the pull up phase is that the deviations in propulsion system performance on both the pull up maneuver and the early part of Stage 2 are large compared to the off nominal deviations sensed during the pull up maneuver. The predictions on which an optimization would be based are therefore not accurate enough to produce an effective improvement in the efficiency of the profile.

Second Stage Boost, Parking Orbit Mode

The boost end conditions for the Hohmann transfer to the parking orbit are calculated in the rendezvous mode prediction program for input to the boost guidance program. Three alternatives for guidance to meet these end conditions were considered. The first method consists of an explicit solution for the minimum energy control profile in terms of the solution of the two point boundary value problem of the calculus of variations. One implementation of equations for this problem is documented in Reference 14. It is included among the guidance modes of the Dyna Soar Boost Simulation program which was used to study the application of the method to the problem.

The second method of boost guidance for the parking orbit mode consists of a nominal attitude-velocity control profile over about two-thirds of the boost time followed by a proportional control law on the last part of the profile. The steering command is proportional to the cross plane velocity-to-be-gained; the pitch attitude command is proportional to the vertical velocity-to-be-gained; and the thrust cut off time is predicted in an iterative loop as a function of the total velocity-to-be-gained.

The third method of boost guidance for the parking orbit mode consists of open loop control of a fixed attitude rate profile. Attitude reference is provided by three strapped down gyros and thrust acceleration is measured and integrated to obtain total velocity impulse. Thrust is cut off when the measured ΔV attains a specified value. This approach would require a minimum stage 2 computational capability for the launch phases of operation.

Second Stage Boost Guidance, Direct Ascent Mode

The initial boost end conditions are predicted by the guidance mode selection program. If there is a small lag or lead at staging, the selected mode is direct ascent. The guidance methods for injection into a direct ascent transfer to rendezvous are essentially the same as for injection into a Hohmann transfer to a parking orbit altitude. The difference is in the calculation of desired boost end conditions.

The boost end conditions are estimated and iteratively corrected in a program which calculates the velocity vector required at the predicted position and time of cut off to meet the requirements for intercepting the target. This prediction continues through and after staging. After staging, the predicted end conditions are inputs to either of the first two boost guidance programs described above for boost to a parking orbit. The end condition predictor - corrector and the current control command program operate sequentially on each iteration. This is a sub-optimal procedure from the theoretical point of view in that the solution for end conditions and control profile are not solved for simultaneously. However, over small deviations of the profile from Hohmann, the iterative process described converges to essentially the same solution that would be produced by a more general optimization. The simplification allows the identical control profile program to be used in the direct ascent and parking orbit modes.

It is unlikely that an open loop control for direct ascent could be successfully implemented because of error effects. However, an analysis was made for this type of control on the assumption that adaptations for time deviations from nominal could be implemented by recalculation of the attitude rate profile before staging.

Ideal Orbital Guidance From Parking Orbit to Rendezvous Transfer Orbit

After injection into the initial coast ellipse, the relative position of the vehicle and target are predicted using Keplerian orbital equations.

The velocity vector and corresponding ΔV required for intercept with the target from a moving point 90° down range is computed continuously. The equations are essentially the same as those required for predicting end conditions on the boost profile. If the solution passes through the conditions for a Hohmann transfer to rendezvous at less than 181° down range from the position of boost cut off, the program mode is switched to generate and control the commands for injection to the rendezvous transfer ellipse at that point. If the conditions for a Hohmann transfer are not met at 181°, the program mode is switched to generate the commands for injection into parking orbit at the coast ellipse apogee. After injection into parking orbit the program is switched back to the mode for seeking the point at which the conditions for a Hohmann transfer to rendezvous are met.

The target position relative to the vehicle, required for prediction of the orbital maneuvers, also defines the target seeker search control profile. The search commands are produced continuously after boost until the target is acquired.

Open Loop Orbital Guidance, Parking Orbit Mode

The simplest form of orbital guidance consists of a fixed event-time profile for control of the sequence of maneuvers to inject into the parking and rendezvous transfer orbits. The parameters for the sequence of events would be predicted by the Stage 1 system before separation. The method would be used in conjunction with open loop boost guidance.

Terminal Maneuver Guidance

The rendezvous terminal guidance is based primarily on the terminal sensor measurements. In the ideal guidance system, the measurements are processed in a dynamic model using the same basic equations required for pre-acquisition orbital prediction and guidance. The difference between predicted and observed measurements (residuals) are processed statistically to produce an optimum solution for the terminal rendezvous maneuver. Computer requirements were estimated for the ideal model and for two simpler models. The simplifications consist of sub-optimal treatment of the target sensor data and neglect of centripetal acceleration in the dynamic model.

Stage 1 Deceleration After Separation

The deceleration to cruise conditions is governed by a predetermined pressure altitude-velocity profile with fuel weight deviation from nominal as a parameter. Further study is required to specify the form of the guidance law, but it is not expected that this will have a significant effect on computer requirements.

2.4.5 Navigation System Equations

Navigation computations fall into two categories: integration of the equations of motion and updating of system parameters based upon sensor data.

Equations of Motion

In inertial systems, thrust and aerodynamic forces, measured by accelerometers mounted on a stable platform, are summed with gravitational forces and integrated to obtain velocity and position in the inertial reference system. In doppler systems, the instantaneous forward velocity of the vehicle relative to the ground is measured by a radar system and resolved into North and East components through a heading reference. The components of velocity are integrated to obtain geocentric position in latitude and longitude. In air-data systems the instantaneous forward velocity of the vehicle relative to the air mass is computed as a function of pressure and resolved into North-East components through the heading reference. The velocity relative to the air mass is combined with estimated wind velocities and integrated to obtain geocentric position in latitude and longitude.

Updating

Until recently, in-flight navigation system updating consisted primarily of two straightforward deterministic processes: position fixing based on star, horizon, and landmark positions, and inertial platform reference axes alignment based on star angular positions. With the increased capability of computers, a much more general approach to system updating becomes feasible, namely, application of optimal statistical estimation to update the total system state. Two forms of the concept were considered in the estimates of computer requirements:

(1) in-flight differential correction of Stage 1 or Stage 2 position and velocity, and inertial reference system alignment and calibration constants based on external observed data (star,omega,landmark);

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(2) alignment and calibration of Stage 2 inertial reference system with the Stage 1 system as master reference; without external references in an in-flight mode; with optical reference in pre-flight alignment and calibration mode.

2.4.6 Computer Requirements Study

Estimated memory requirements for the individual navigation and guidance system functions outlined above are given in Table 2.4-1. The estimates are based on work performed on a number of projects, including the Dyna Soar vehicle computer work, the Saturn V Launch Vehicle Guidance and Navigation functional description, and computer trade studies on the AGM-69 program. It is assumed that there is one instruction per word of memory, with the instruction word and data word of equal size. The word length is 16 bits and the computers have double-precision program capability. Tables 2.4-2 and 2.4-3 list the total computer memory requirements for Stage 1 guidance and navigation for the rendezvous and cruise missions, respectively. Table 2.4-4 similarly lists the requirements for Stage 2 on the rendezvous mission.

The memory requirements for minimum, medium, and maximum capability systems, quantized in terms of 2048-word modules, are in the ratio of 3:4:5 for Stage 1 on the rendezvous mission; 2:3:5 for the (5000 n.m.) 9,260 km cruise mission; and 1:2:3 for Stage 2 on the rendezvous mission. The transformation of these memory size ratios to cost and weight ratios depend on the correlated computation speed requirements because memory costs are a function of memory cycle time, which in turn is an important factor in computation speed. A preliminary analysis of the transformation factors follows.

The increase in functional capability from medium complexity to maximum complexity on Stage 1 is not reflected in a greater computer speed requirement. This is because the updating technique performed during the cruise phase in conjunction with navigation does not have to be executed in real time and may be performed at a relatively low repetition rate compared to the integration of the equations of motion. Similarly, the mode control functions added in going from minimum to medium capability place very low demands on computation speed. The principal difference in speed requirements between the minimum system and medium system is in the integration of the equations of motion. The inertial navigation requires higher computing rate than the doppler or air-data systems.

Again, on the second stage, the principal increase in computer speed requirements in going from the minimum to the maximum system is in the integration of the equations of motion on the boost phase. The added mode capabilities, the ideal orbital equations, and the sensor data processing in the terminal phase are complex in form, but may be executed at low repetition rates.

Summarizing, it is estimated that a computation speed characterized by add and multiplication times of 12μs and 50μs, respectively, (i.e., a medium speed computer in terms of present state of the art) will be adequate for the medium and maximum capability systems for both Stage 1 and Stage 2. A lower speed, such that add and multiplication times are respectively 30 μs and 250 μs, will be adequate for the minimum capability systems.

Requirements for capability as defined above in terms of memory and computation speed must be converted to relative weight, reliability, and cost ratios for the purposes of optimization of system specifications.

Reliability and cost data are presented in Sections 2.6 and 2.7. The weight factors divide into two parts: the weight factor proportional to memory size requirements, and the weight factor dependent on computing speed requirements. For the purposes of the present analysis it is assumed that the relative weights of the computer systems are proportional to the relative weights of the memory systems. Assuming that the central computer and the memory read-write control circuits combined are equal in weight to two 2048 word modules, the relative system weights are as tabulated in Table 2.4-5. For the rendezvous mission, there is a 7.3 kg (16-pound) weight variation from minimum to maximum requirements on both Stage 1 and Stage 2. Using an exchange ratio of 9:1 of Stage 1 weight to payload weight, the total payload weight penalty in going from a minimum to maximum capability is 7.3 kg (16 pounds). On the 9,260 km (5000 n.m.) cruise mission, the penalty is 11.8 kg (26 pounds).

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TABLE 2.4-1 COMPUTER PROGRAM INSTRUCTION REQUIREMENTS VERSUS FUNCTION

Program Functions	No. of Instructions Stage 1	Stage 2	Comments (Whole Number Computers)
Master Control Routine	300	150	
Arithmetic Subroutines	400	200	Stage 1: Sine, Cosine, Arc Tan, Square Root, Log, Exponential, Table Lookup Stage 2: Sine, Cosine, Square Root, Arc Tan
Computer Loading	150	50	
Telemetry Input-Output	300		Stage 2 may not require telemetry, given hard lines to Stage 1.
Nav. & Guid. System Checkout			
Preflight Countdown			
• Autonomous or	1300	600	Alternatives marked with dots are mutually exclusive
• Ground-Monitored	600	300	
In Flight Error Detection	300	150	This program operates after N & G preflight countdown; the 300 words of Stage 1 are included in the 1300 words of the autonomous mode countdown.
Navigation			
• Dead Reckoning Using Dynamic Pressure Sensors, Magnetically-Damped Gyro Compass, and Altimeter	250	Not Applicable	First indented systems are mutually exclusive
• Doppler Navigation With Magnetically Damped Gyro-Compass	250	Not Applicable	
• Inertial System Dead Reckoning			
• Gimballed Platform	1000	800	
• Strapped Down Platform	--	1000	Strapped down platform includes a Digital Diff. Analyzer for Euler Angle rate generation.

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TABLE 2.4-1 (CONTINUED)
COMPUTER PROGRAM INSTRUCTION REQUIREMENTS VERSUS FUNCTION

Program Functions	No. of Instructions Stage 1	Stage 2	Comments (Whole Number Computers)
Navigation, Cont.			
• Stellar - Inertial System	1300	1000	In these modes the low frequency noise of the inertial system is damped with the low frequency star and/or doppler data being the reference.
• Doppler - Inertial System	1250	Not Applicable	
• Stellar-Doppler Inertial System	1500	Not Applicable	The Stage 1 system is Shuler tuned; the Stage 2 system is not.
• Omega Navigation	1500	Not Applicable	
• Inertial system with updating by statistical estimation based on star and/or position fix data of any type.	2000		With this technique gyro drifts, accelerometer biases, and accelerometer scale factor errors are estimated as well as platform alignment and vehicle state vector.
• Stage 2 Inertial System Alignment by Stage 1 System			
• Optical Alignment and direct state vector initialization	200	100	
• Statistical estimation	800	0	The Stage 2 inertial system operates in its normal mode during alignment.
Guidance			
Climb Phase Guidance	150		
Cruise Phase Guidance			
• Nominal Course to Target Plane and Nominal Turn	500		Take-off time + flight plan by ground.
• Time-Adaptive Path to Target Plane	800		Take-off time and flight plan determined in prelaunch mode.
Pre-Launch Guidance Constant Computation for Simplified Second Stage			

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TABLE 2.4-1 (CONTINUED)
COMPUTER PROGRAM INSTRUCTION REQUIREMENTS VERSUS FUNCTION

Program Functions	No. of Instructions Stage 1	No. of Instructions Stage 2	Comments (Whole Number Computers)
Guidance, Cont.			
• Parking Orbit Method	100		
• Direct Ascent	250		The timing of parking orbit injection is modified to compensate for lead time variation. The pitch gyro torquing rates and the cut off conditions of the second stage boost are modified to compensate for lead time variation. The timing of the nominal initial rendezvous maneuver is also modified.
Prelaunch Second Stage Checkout and Range Safety	200	200	Safety checks are monitored on Stage 1.
In-Plane Pull-Up Maneuver Guidance and Second Stage Launch Control	300		Guidance in this phase also includes the steering program in effect on earlier part of turn into target plane.
Return to Cruise Conditions and Return to Base Guidance with or without Rendezvous with Refueling Vehicle.	200		Program mode variations of outbound cruise phase guidance.
Pilot Displays and Controls	500		
Second Stage Boost Guidance for Rendezvous			
• Open Loop Polynomial Forms		200	
• Explicit Adaptive, Not Minimum Fuel		500	
• Minimum Fuel Adaptive		1000	
Second Stage Orbital Orbital Guidance			
• Fixed Injection Thrust Angle and Timing; Fixed Target Search Control		300	

TABLE 2.4-1 (CONTINUED)
COMPUTER PROGRAM INSTRUCTION REQUIREMENTS VERSUS FUNCTION

Program Functions	No. of Instructions Stage 1	Stage 2	Comments (Whole Number Computers)
• Ideal Guidance, Adaptive to Boost End Condition and/or Target Update Information		800	
Terminal Maneuver Guidance			
• Minimum		250	Differential gravity neglected; sensor data not processed optimally.
• Optimum		700	Dynamic Model includes orbit mechanics; minimum variance sensor data processing.

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TABLE 2.4-2

RENDEZVOUS MISSION STAGE 1 COMPUTER MEMORY REQUIREMENTS VERSUS PROGRAM FUNCTIONS SPECIFICATIONS (State of the Art)

Example Program Specifications	Memory Size: Instructions (Words)	Memory Size: Data (Words)	Total in 2048-Word Modules
Minimum Capability	4600	400	3
Data Link With Ground and Stage 2 Ground-Monitored Checkout In-Flight Error Detection Doppler or Air-Data Navigation System, With Updates From Stage 2 System Flight Plan from Ground Cruise Course to Steer: Explicit Geometric Solution, But Limited to Nominal End Conditions Nominal Maneuver into Target Plane Nominal Pull-Up for Parking Orbit Pilot Displays; Range Safety			
	6000	600	4
Autonomous Checkout, But With Ground Data Link Inertial Navigation With No Updating Pre-Launch Calculation of Take-Off Time, and Initial Course to Steer Adaptive Cruise Course to Steer Multi-Mode Rendezvous Launch Control Pre-launch Range Safety Checks Integrated With Stage 2 Pilot Displays; Range Safety			
Maximum Capability Same as Medium Except Capability is Added For Statistical Data Processing:			
Calibrating Stage 2 Inertial System From Stage 1 Inertial System	7200	800	4
Calibrating Both Systems, Using Observed Data	9200	1000	5
Star Only			
Star and Omega and Landmarks			

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TABLE 2.4-3

CRUISE MISSION COMPUTER MEMORY REQUIREMENTS VERSUS PROGRAM FUNCTION SPECIFICATIONS (State-Of-The-Art)

Example Program Specifications	Memory Size Instructions (Words)	Memory Size Data (Words)	Total in 2048-Word Modules
Minimum Capability	3200	200	2
Data Link With Ground			
Ground-Monitored Pre-flight Checkout			
In-Flight Error Detection			
Doppler or Air Data Navigation			
Flight Plan Acceptance and Enroute Changes			
Cruise Course to Steer: Explicit Geometric Solution of Azimuth on Great Circle to Destination; Multiple Destinations			
Medium Capability	4200	300	3
Data Link With Ground			
Autonomous Checkout			
Stellar-Inertial Navigation			
Guidance as Above			
Maximum Capability			
Same as Medium, Except Capability			
For Statistical Data Processing:			
Star Tracking Only	6200	600	4
Star, Omega, and Landmark	7700	800	5

TABLE 2.4-4

RENDEZVOUS MISSION STAGE 2 COMPUTER MEMORY REQUIREMENTS VERSUS PROGRAM FUNCTION SPECIFICATIONS (State of the Art)

Example Program Specifications	Memory Size Instructions (Words)	Data (Words)	Total in 2048 Word Modules
Minimum Capability	1800	150	1
Data Link With Ground and Stage 1 In-Flight Checkout Monitored By Stage 1 Nominal Parking Orbit Profile Control With Parameters Inserted by Stage 1 Before Separation No Navigation System (Open Loop) Minimum Terminal Guidance			
Medium Capability	3500	300	2
Data Link and Checkout as Above Inertial Navigation Multi-Mode Rendezvous With Boost Non-Optimum, But Explicit Solutions for End Conditions Ideal Orbital Guidance and Search Control Minimum Terminal Manuever Guidance			
Maximum Capability	4500	500	3
Same as Medium, Except: Minimum Fuel Boost Guidance Optimum Terminal Rendezvous Maneuver Data Processing and Guidance			

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TABLE 2.4-5

COMPUTER WEIGHT VERSUS CAPABILITY (Stage-of-the-Art)

Application	Capability	Relative Weight Ratio	Estimated Actual Weight	
Stage 1, Rendezvous Mission	Minimum	5	18 kg	(40 lbs.)
	Medium	6	22	(48)
	Maximum	7	25.4	(56)
Stage 1, 5000 n.m. Cruise Mission	Minimum	4	14.5	(32)
	Medium	5	18	(40)
	Maximum	7	25.4	(56)
Stage 2, Rendez-vous Mission	Minimum	3	11	(24)
	Medium	4	14.5	(32)
	Maximum	5	18	(40)

2.5 Payload Performance Penalties

2.5.1 Penalties Due to Stage Timing Errors

Payload performance as previously defined is measured in terms of weight and targeting accuracy at second stage launch. The errors accrued during launch vehicle operation due to dispersion in atmosphere, aerodynamics and propulsion can be related to time and position and, corrected in three ways: (1) during launch-vehicle cruise, (2) in direct ascent rendezvous, and (3) by the parking orbit method. During the Stage 1 cruise phase, position with respect to the target plane can be continuously monitored and used to alter the cruise course to correspond to updated prediction of staging position and time. In the direct ascent rendezvous method, timing errors are compensated by altering the second stage boost trajectory to terminate at the desired end conditions. The parking orbit method requires the vehicle to be launched into a parking orbit for a catch-up period prior to transfer to the final orbit.

Stage Timing Errors

Dispersions in vehicular characteristics and atmospheric parameters result in off-nominal range, time and fuel usage. They are generally interrelated. When range is short, time and fuel generally are also short such that when range is made up, time and fuel are compensated. For determination of the three sigma timing error associated with the first stage this compensation was considered. Figure 2.5-1 is a table of errors in fuel and time corrected for range resulting from major dispersions throughout the launch mission. These data indicate three sigma stage timing error of 56.2 seconds during the acceleration phase. The major timing error during cruise/turn originates from unpredicted wind since this is the only factor influencing range rate for a perfect guidance system. The root sum square of this latter effort with the three sigma climb/acceleration error brings the cumulative timing error to slightly over 60 seconds.

2.5.1.1 Correction During Launch Vehicle Operation

Four methods of correcting stage timing errors during first stage operation have been investigated. They have been named the lead time method, the trombone turn method, the adaptive cruise method and throttle control. Numerical analysis of the first three methods are given in Appendix A2. Sketches are shown in Figure 2.4-5. The lead time method allows a nominal delay in staging after the launch vehicle ground track is established in the target plane as shown in Figure 2.4-5. The nominal delay is equal to the expected minus value of the three sigma error. Additional fuel is allowed for plus three sigma error. The trombone turn method delays or accelerates the staging time and at the same time moves the staging point down range or uprange from the nominal as shown in Figure 2.4-5. This is accomplished by off-nominal turn courses in the appropriate direction. The adaptive cruise method produces similar results as the trombone turn method but is accomplished by off-nominal cruise heading as sketched in Figure 2.4-5. Throttle control can also be used for timing error correction. As in the lead time method, a fuel penalty is associated with flying the nominal. This is because the most efficient cruise occurs at the maximum velocity. In order to provide correction capability for lead

Figure 2.5-1

LAUNCH VEHICLE DISPERSION IN TIME AND FUEL 3704 km (2000 NM) OFFSET MISSION

Dispersion Source	Value of Dispersion	Acceleration Phase Time Sec.	Acceleration Phase Fuel (Pounds)	Acceleration Phase kg	Outbound Cruise Time Sec.	Outbound Cruise Fuel (Pounds)	Outbound Cruise kg	Return Phase Time Sec.	Return Phase Fuel (Pounds)	Return Phase kg
Lift	5%	2.4	(143)	65	0	(2830)	1284	0	(1600)	725
Drag	5%	27.3	(1826)	828	0	(2830)	1284	0	(1600)	725
Thrust	5%	41.6	(802)	364	0	(2400)	1090	0	(1360)	616
Fuel Flow	5%	1.9	(2472)	1120	0	(2830)	1284	0	(1600)	725
Weight	1%	7.5	(622)	282	0	(565)	256	0	(320)	145
Atmosphere	*	24.14	(368)	167	0	(1410)	640	0	(800)	363
Wind	*	*	*	*	21.8	(833)	378	57.3	(2270)	1030
RSS		56.2	(3260)	1480	21.8	(5720)	2590	57.3	(3930)	1780
Cumulative RSS Dispersion		56.2	(3260)	1480	60.2	(6580)	2980	83.2	(7660)	3480

*Atmospheric density and wind assumed predictable to equivalent tapeline altitude of 305 meters (1000 feet) for acceleration phase. Wind during remainder of mission from Handbook of Geophysics, 1960.

or lag the nominal cruise velocity must be less than maximum. The first stage fuel penalty is defined as the fuel required to follow the nominal flight path plus the extra fuel required to correct $\pm$ 3 sigma errors. First stage fuel penalties are tabulated below for the four methods. They represent timing error correction only and do not include the three sigma fuel penalties accrued during off-nominal flight which are shown in Figure 2.5-1.

Payload Penalties for One Minute Timing Error

	Nominal Flight Path				Error Correction			
	Stage 1 Fuel		Orbital Payload		Stage 1 Fuel		Orbital Payload	
Method	(lbs.)	kg	(lbs.)	kg	(lbs)	kg	(lbs.)	kg
Lead Time	(6050)	2740	(660)	299	(12350)	5,600	(1350)	612
Throttle Control	(690)	312	(76)	34	(1390)	630	(151)	69
Trombone Turn	(80)	36	(9)	4	(850)	385	(93)	42
Adaptive Cruise	0	0	0	0	(500)	226	(55)	25

The first stage fuel penalty is assumed to trade equally with Stage 2 launch weight penalty. The derivation of this exchange ratio is based on a constant structural relation,

$$\lambda' = \frac{W_P}{W_P + W_I} = \text{Constant}$$

where W_P = propellant weight
W_I = inert or structural weight

which is derived in Appendix A1.

Penalties for off-nominal cruise courses include the effect on return fuel but throttle control is used only on the outbound leg. But it is a compensating effect for the trombone turn and adaptive cruise methods. The magnitude of the stage timing errors and time when they are detected has a significant influence on the selection of a correction method. If they occur early in the trajectory, the adaptive cruise method becomes attractive because of the smaller deviation from the minimum fuel flight path. However if large errors occur in the cruise phase, the trombone method may well be the selected scheme. Preliminary analysis indicates large timing errors are possible in the acceleration phase. For example, stage time errors are quite sensitive to deviations in thrust, drag and atmospheric effects during acceleration. On the other hand, winds have the only dominating influence during the cruise/turn. In practice, a combination of all three course alternate methods would probably be used because no greater guidance complexity is added to the system by the addition of these simple equations than is required to determine the position of the moving target plane under nominal flight path conditions for example. This is particularly true if direct ascent is desired because the lead time method would allow final corrections. The throttle control method is always available but would probably not be considered as a design method because of the reduced aerodynamic efficiency of operating at off-design Mach number. The study indicates incremental velocities in excess of 150 m/sec. (500 fps) to correct for 60 seconds timing error.

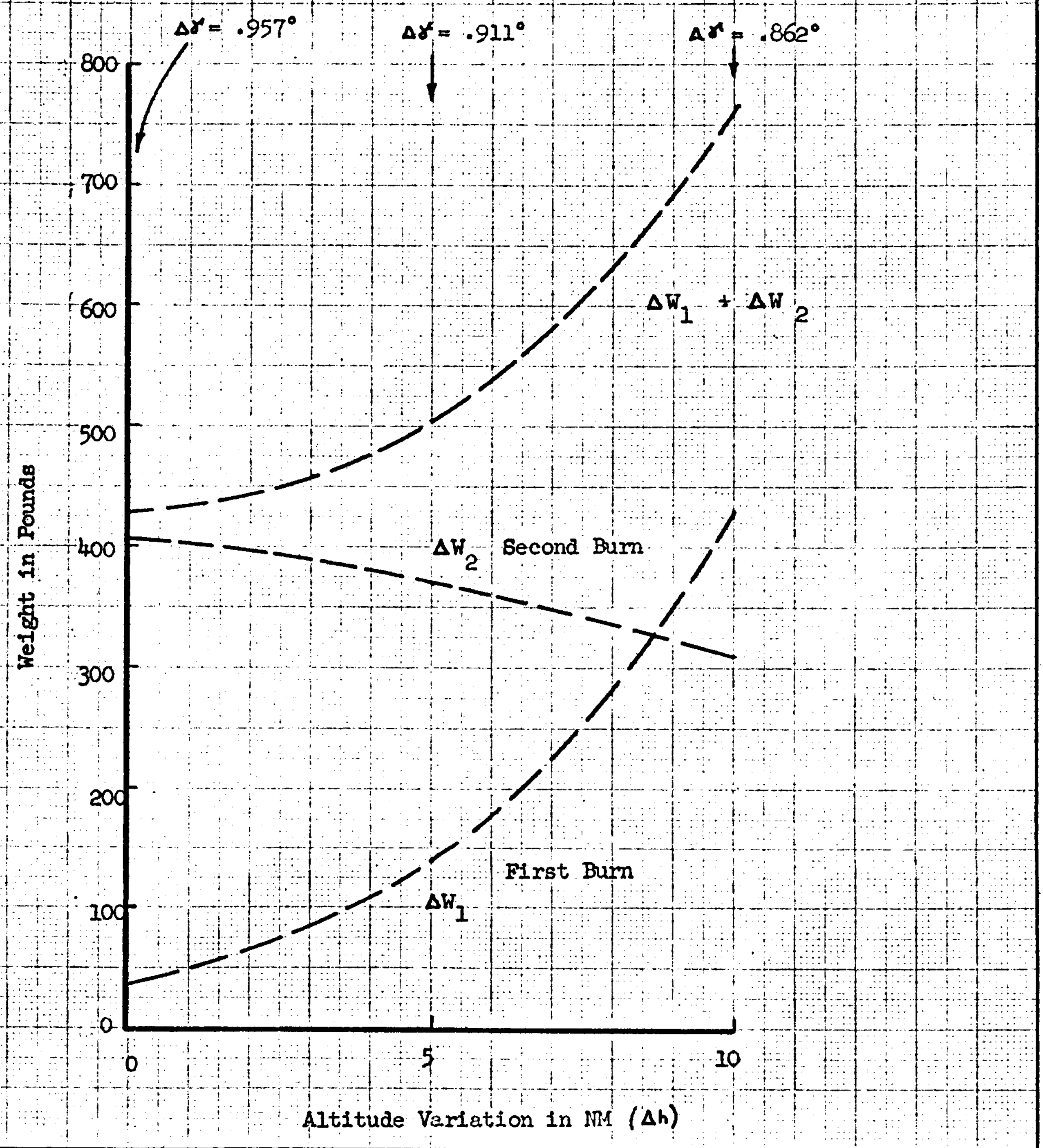
If indeed the combination of course alteration methods were used, then acceleration phase errors as well as early cruise errors could be nulled by the adaptive cruise method, cruise time errors could be nulled by the trombone turn method and final phasing errors could be corrected by the lead time method. This allows the large stage time errors to be corrected by the most economical method.

2.5.1.2 Correction During Second Stage Boost

The rendezvous payload penalties for non-ideal timing at second stage launch were estimated from data obtained on runs using the minimum fuel guidance law in the Dyna Soar Boost Simulation program. The specified second stage end conditions for the first thrust period were modified to achieve rendezvous with launch time deviations up to one minute. Combinations of altitude and flight path angle variations to meet the constraints were studied. It was found that the payload penalty to meet off-nominal intercept conditions are substantially lower when only the flight path angle at boost cut off is varied. This is shown in Figure 2.5.2 in which the payload penalties are plotted versus a combination of boost cut off altitude and flight path angle variations. The total penalty is minimized when the altitude variation is zero. Figure 2.5.3 shows the payload penalty versus time deviations when flight path angle only at boost cut off is modified to meet intercept conditions. The penalty for a twenty second deviation is 25 kg (56 pounds). The penalty for a one minute deviation is 197 kg (435 pounds).

2.5.1.3 Correction By Parking Orbit Method

The parking orbit method involves boosting the payload to an orbit below that of the target vehicle and providing time for a slow catch up before transferring to the target orbit. This is the most efficient method and would probably be used if time is not a critical factor. For example, the propellant difference between a direct ascent and interim use of a parking orbit 185 km (100 nautical miles) below the target orbit is 23 kg (50 pounds). This exchanges with 25.6 kg (56.5 pounds) of payload because of the additional inerts. This analysis does not consider errors associated with guidance during long periods in parking orbit, analyzed in Section 2.5.2.



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC					RENDEZVOUS FUEL PENALTY DUE TO BOOST END CONDITION VARIATIONS	FIGURE 2.5-2
CHECK						
APPD.						
APPD.						

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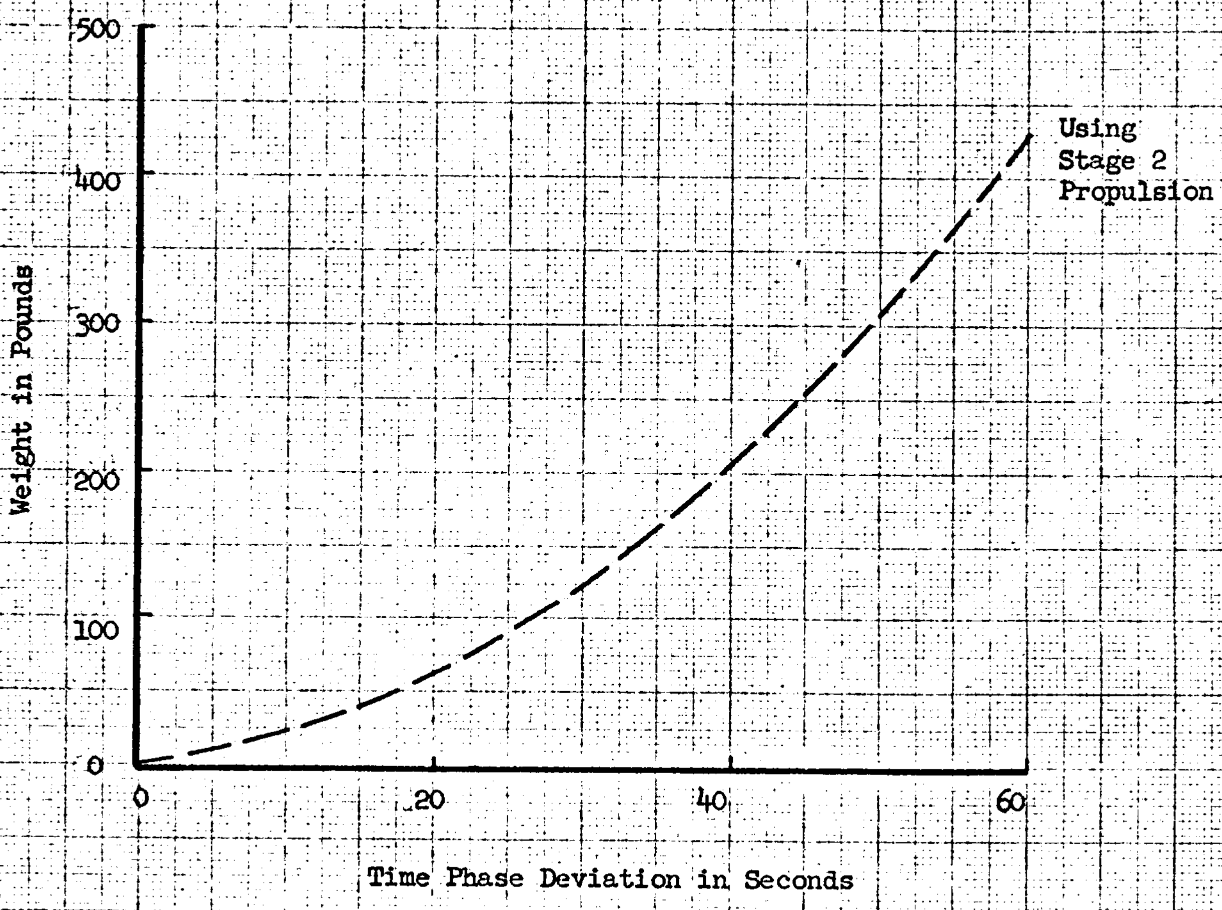
Notes:

1. Data based on direct ascent, ideal guidance



	INITIALS	DATE	REV BY INITIAL	DATE	TITLE	MODEL
CALC					PAYLOAD PENALTY DUE TO TIME PHASE VARIATION	FIGURE 2.5-3
CHECK						
APPD.						
APPD.						

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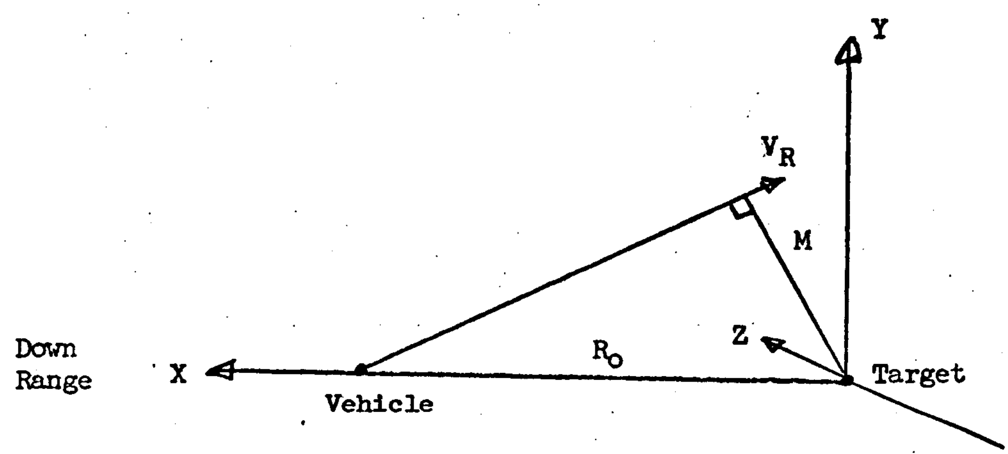
2.5.2 Payload Weight Penalties Due to Navigation System Errors

Transformation of Navigation Errors to Rendezvous ΔV Penalty

The rendezvous ΔV penalty is a function of the rendezvous maneuver technique as well as of the trajectory errors at the start of the terminal maneuver. For purposes of comparison, a simplified model of the transformation of trajectory errors to ΔV penalty was used. Figure 2.5-5 shows the target-vehicle relative geometry, neglecting gravity acceleration, at target acquisition range, R_o. The navigation errors, propagated to the rendezvous point, are represented by the error vector (ΔX, ΔY, ΔZ, $\Delta \dot{X}$, $\Delta \dot{Y}$, $\Delta \dot{Z}$).

Figure 2.5-5



The rendezvous maneuver consists of two impulses. The first impulse places the vehicle on a collision course with the target after establishing the relative track angle. The second impulse establishes the rendezvous conditions of zero relative velocity at intercept. The first impulse, ΔV_1, is proportional to the transverse error and inversely proportional to the total closing time, t, as follows:

$$\Delta V_1 = \Delta Y^2 + \Delta Z^2 \; / \; \Delta t$$

The second impulse is a function of the nominal closing velocity increment, ΔV_{nom}, and the trajectory velocity deviations from nominal at intercept. In addition there is a penalty associated with down range position error. In order to ensure a relative geometry at acquisition of the type shown in Figure 2.5-5, the nominal aim point is placed down range and high in altitude. The ΔV penalty associated with this is due principally to the development of a nominal non-zero flight path angle which must be corrected on the maneuver. It was assumed that this penalty is proportional to the down range error. The proportionality factor, K, was obtained by solving a case with a median down range error. It is equal to 0.0005 m/sec/m (ft/sec/ft), and was the same on the terminal geometry for Hohmann transfers from 74 km (40 N.M.) altitude and 298 km (161 NM) altitude, representing respectively the modes of direct ascent and parking orbit. The penalty on the second impulse is given by the expression:

$$\Delta V_2 = \Delta V^2_{Nom} + K^2 \Delta X^2 + \Delta V_1^{\,2} + \Delta \dot{X}^2 + \Delta \dot{Y}^2 + \Delta Z^2 - \Delta V_{Nom}$$

Finally, the total penalty is the root sum square of ΔV_1 and ΔV_2.

Results

Three-sigma rendezvous maneuver ΔV penalties for a representative set of the systems defined in paragraph 2.3.3 are listed in Figure 2.5-6. The systems selected are in the class of all-inertial systems, with accuracy performance dependent only on the second stage navigation system. The list includes a direct ascent case and a parking orbit case with two terminal closing times of 10 minutes and 20 minutes on each. The direct ascent case is for a perfectly timed launch of the second stage. A launch lag time of 2.5 minutes is compensated on the example parking orbit case with the parking altitude at 298 km (161 N.M.), 74 km (100 N.M.) below the target altitude. The 2.5 minutes lag time is at the five-sigma level of expected stage and flight time uncertainty, and thus may be considered an upper limit on time to be compensated. The greater penalty for the parking orbit as compared with the perfectly timed direct ascent is due primarily to a greater sensitivity of cross plane position error at rendezvous to cross plane velocity error at boost cutoff.

The ΔV penalties for purely navigation system errors are equal to the penalties listed for the perfectly timed direct ascent. Given a closing time of 20 minutes, the penalties are 61 m/sec, 18 m/sec, and 4.6 m/sec. (200 FPS, 60 FPS, and 15 FPS), respectively, for the low, medium and high accuracy navigation systems.

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Figure 2.5-6
TERMINAL RENDEZVOUS ΔV (3)
PENALTIES FOR BOOST GUIDANCE SYSTEM ERRORS AND TARGET PREDICTION UNCERTAINTY

Closing Time	System # and Accuracy Level		Rendezvous ΔV and Payload Weight Penalties FPS and Pounds							
			Direct Ascent (180° Coast)				Parking Orbit (450% Coast Angle)			
			m/sec.		kg		m/sec		kg	
10 Minutes	4	Low	134	(440 FPS)	449	(990 lbs.)	174	(570 FPS)		(1280 lbs.)
	6	Medium	70	(230)	236	(520)	64	(210)	213	(470)
	7	Medium +	39.6	(130)	131	(290)	55	(180)	181	(400)
	10	High -	21.4	(70)	72.5	(160)	24.4	(80)	81.5	(180)
	13	High +	9.2	(30)	31.8	(70)	10.7	(35)	36.3	(80)
20 Minutes	4	Low +	61	(200)	204	(450)	91.5	(300)	306	(675)
	6	Medium	30.5	(100)	102	(225)	45.7	(150)	154	(340)
	7	Medium +	18.4	(60)	61.2	(135)	36.6	(120)	122	(270)
	10	High -	9.2	(30)	30.8	(68)	10.7	(35)	36.3	(80)
	13	High +	4.6	(15)	15.4	(34)	6.1	(20)	20.4	(45)

At an exchange ratio of 3.35 kg per MPS (2.25 pounds per FPS), the corresponding payload weight penalties are respectively 204, 61, and 15.4 kg (450, 135, and 34 pounds).

The difference between the ΔV penalty for the direct ascent and parking orbit are due to the propagation of navigation errors on the parking orbit. They may be viewed as part of the penalty for compensating for stage timing errors. These penalties are 30.5, 15.2, 1.5 m/sec (100 FPS, 50 FPS, and FPS), respectively, for the low, medium, and high accuracy systems. The corresponding payload weight penalties are 102, 34.2, and 17.2 kg (225, 112, and 38 pounds). These are compared in paragraph 2.5.3 with the penalties for stage timing corrections in the direct ascent mode.

2.5.3 Comparison of Direct Ascent and Parking Orbit Weight Penalties Due to Time Errors

The primary performance penalty associated with the selection of the form of the guidance law and corresponding vehicle operating mode is due to time variations. The factors that cause the time variations can not be changed directly by the guidance systems; however, these factors establish requirements on the guidance system. The two guidance modes - direct ascent and parking orbit modes - are compared in Figure 2.5-7 for the payload weight penalty required to correct for time variations.

The total time error to be corrected is the same for both modes, 2.5 minutes. In the direct ascent mode it is assumed that most of the time error (2.2 minutes) is corrected in the cruise phase with the adaptive cruise heading mode and that 20 seconds of residual delay is corrected in the second stage thrust period with a minimum fuel guidance law. In the parking orbit mode the total time error of 2.5 minutes is corrected in the parking orbit. The payload penalties for the time error are a function of the Stage 2 navigation system accuracy as explained in 2.5.2. Only the excess penalty compared to the corresponding direct ascent case are included; the penalties due to navigation errors have not been included here, see 2.5.2. The Figure 2.5-7 results represent the timing error value at which mode switching from direct ascent to a parking orbit mode is desirable.

2.5.4 Stage 1 Performance Trades on Return Phase

Three factors influence the performance requirements of the Stage 1 navigation system independent of the Stage 1 navigation and guidance selection. The first factor is the fuel penalty to correct return phase navigation errors. The second factor is the requirement for a refueling rendezvous with a tanker on the return phase. The third factor is the 9,260 km (5,000 N.M.) cruise mission.

Assuming that an accurate Stage 2 system is used for the outbound phase, the effect of navigation errors on the return phase will be considered. Radio aids such as VOR and Tacan will be used to locate the landing site as it is approached. It is assumed that the effective range of these aids is 185 km (100 N.M.). A 18.5 km/hour (10 N.M./hour) CEP Stage 1

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Figure 2.5-7 Summary of Weight Penalties for Direct Ascent and Parking Orbit Modes of Correcting Time Sensors

Mode	Time Error, Minutes: Before Staging	Time Error, Minutes: After Staging	Payload Penalty, kg (Pounds): Before Staging	Payload Penalty, kg (Pounds): After Staging	Payload Penalty, kg (Pounds): Total
Direct Ascent					
Adaptive Cruise	2.2		54.5 (120)		
Minimum Fuel Stage 2 Guidance		0.3		22.7 (50)	77 (170)
Parking Orbit					
20 Minute Rendezvous Closing Time					
Navigation System Accuracy					
Low		2.5	0	102 (225)	102 (225)
Medium		2.5	0	50.8 (112)	50.8 (112)
High		2.5	0	17.3 (38)	17.3 (38)

navigation system will require fuel to correct a 55.5 km (30 N.M.) error. The nominal return time is 53 minutes and a fuel margin for at least three sigma errors is required. The along track component of error when descent is started 55.5 km (30 N.M.) too soon is the most expensive to correct. The subsonic range fuel trade is 10.3 kg/km (42.0 lbs/N.M.) and the supersonic trade is 4.5 kg/km (18.4 lbs/N.M.). The difference 5.76 kg/km (23.5 lbs/N.M.) is the fuel penalty sensitivity to correct the navigation error. A Stage 1 fuel margin of 320 kg (705 lbs) is required. This is equivalent to 35 kg (77 lbs) payload with a 4.15 kg (9.16 lb.) exchange ratio (see Appendix A-1). The payload penalty for a 1.85 km/hr. (1 N.M./Hr.). Stage 1 system is 3.6 kg (8 lbs.) and for a 0.185 km/hr (0.1 N.M./hr) system it is 0.36 kg (0.8 lbs).

The fuel penalties associated with correcting navigation errors during a refueling rendezvous that is accomplished at subsonic speed, as specified, would have similar excess fuel requirements. However, since the refueling will take place near the staging point, the outbound phase errors are expected to be most significant. Thus, the refueling rendezvous requirement is not expected to be decisive in a tradeoff for selecting the Stage 1 system.

The nominal mission time for the 9,260 km (5,000 N.M.) cruise mission is about 1.7 hours. Thus, the penalties for correcting terminal navigation errors for the cruise mission are almost twice the penalty for the rendezvous mission return phase discussed above.

The overall effect of payload penalties as a function of accuracy performance is presented in the tradeoffs Section, 2.8. The computer cost studies and payload penalty considerations indicate it is desirable to include a multi-mode capability so that the most effective mode can be chosen for the specific operational situation of a mission.

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2.6 RELIABILITY ANALYSIS

2.6.1 Introduction and Purpose

This section summarizes the reliability work completed during the Phase I study. The primary purpose of this effort was to provide reliability information in support of trade studies to select two promising concepts for detailed study in Phase 2.

The reliability effort was based on the premise reliability is inherently sensitive to primary operational variables such as weight, cost, and accuracy and, is therefore a dynamic, rather than a static parameter. Accordingly, reliability support was provided through the implementation of the following tasks:

a) Establishment of reliability prediction procedures and assumptions to be employed in the study,

b) Compilation of failure rate data on all levels of guidance and navigation equipments, i.e., systems, subsystems and components; and,

c) Determination of reliability feasibility ranges for the equipments.

Subsequent paragraphs discuss the results from these reliability activities.

2.6.2 Summary

The inherent mission reliability feasibility range for a non-redundant navigation and guidance system concept as proposed in the study contract was derived to be 0.9591 to 0.9966, with a corresponding MTBF range of 236 to 3015 hours; for a completely redundant system the reliability range is 0.9991 to 0.999994. The reliability feasibility range of the non-redundant navigation and guidance system for the first stage is 0.9787 to 0.9986, with a corresponding range of 366 to 5450 hours MTBF; for the non-redundant second stage the reliability range is 0.9800 to 0.9980, with a MTBF range of 655 hours to 6750 hours. Tables 2.6-1 and 2.6-3 present the data that were used to derive these reliability ranges.

These results indicate that study mission success objective of no less than 0.95, initially with a design objective of 0.99 or better are realistic and feasible goals for the navigation and guidance system concept proposed in the study contract.

2.6.3 General Approach

This section describes the general approach employed to provide reliability support during the study.

2.6.3.1 Reliability Prediction Procedure and Assumptions

The procedure employed in performing reliability predictions involved the following

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TABLE 2.6-1

SYSTEM RELIABILITY FEASIBILITY

System Configuration	Stage	Reliability Range					
		Nonredundant				Redundant	
		MTBF (Hrs)		Reliability		Reliability	
		Low	High	Low	High	Low	High
1. Platform + Computer + Displays	1	555	5,450	.9860	.9986	.99981	.999998
2. Platform + Computer	2	715	6,750	.9816	.9980	.99966	.999996
3. Stellar Inertial + Computer + Displays	1	525	5,180	.9852	.9985	.99978	.999998
4. Stellar Inertial + Computer	2	665	6,250	.9800	.9979	.99960	.999996
5. Doppler Inertial + Computer + Displays	1	380	3,250	.9795	.9976	.99958	.999995
6. Stellar Doppler Inertial + Computer + Displays	1	366	3,140	.9787	.9975	.99955	.999994

TABLE 2.6-2

RELIABILITY FEASIBILITY RANGE

Subsystems	MTBF Hours Low	MTBF Hours High
1. Inertial Platform and Platform Electronics	1,000	10,000
2. Star Tracker	10,000	100,000
3. Digital Computer	2,500	20,000
4. Controls and Displays	2,500	30,000
5. Doppler Radar	1,200	8,000

TABLE 2.6-3

TYPICAL MISSION PROFILE DATA

Phase	Operating Time	Environment K-Factor	Operating Time X K-Factor (Equiv. Mission Time
Stage 1			
Cruise out	41.0 minutes	5.0	
Return	52.8 "	5.0	7.8 hours
Stage 2			
Cruise out	41.0 minutes	5.0	
Thrust period	4.4 "	125.0	
Coast	45.0 "	1.0	13.3 hours

basic steps:

1) Define the system
2) Establish reliability model
3) Determine part population for each functional block
4) Determine appropriate stress factors for each part
5) Assign applicable failure rates to each part
6) Compute subsystem reliability
7) Compute overall system reliability.

The first step includes the following elements:

1) Determine purpose or intended use; i.e., define the mission(s)
2) Determine conditions which constitute system failure
3) Determine functional and physical boundaries of system.

The second step involves tasks such as:

1) Construction of a reliability block diagram to the lowest identifiable function, showing the relationships necessary for successful system operation, and clearly indicating alternate modes of operation.
2) Establishment of mathematical equation of reliability for the system and each functional block in step 1) above.

Steps 3, 4 and 5 required the determination of the part population for each functional block, and the stress factors and failure rate for each part in each functional block. Since the nature of this study precluded making these determinations at the functional and part levels, the stress factors and failure rates were derived for the equipments at the subsystem and system levels. These data were based on data from existing systems having functional and operational characteristics similar to those of the system defined in the study contract.

The sixth and seventh steps entail the computing and combining of reliabilities of lower levels of assemblies to obtain subsystem reliabilities and the overall system reliability.

Standard reliability engineering techniques were used to accomplish these steps. Also, the following assumptions were used:

1) The reliability distribution is exponential, namely $R = e^{-\lambda t}$, except where otherwise noted. This means that all parts are assumed to have a constant failure rate within a particular environmental envelope.

2) Except as noted, system elements (i.e., parts, components, etc.) are assumed to be related in a series manner such that failure of each element are considered to be independent and that failure of any element will cause system failure. The model for this arrangement of system elements is as follows:

$R_o = \prod_{i=1}^{n} R_i$ where R_o is the overall system reliability and R_i is the

reliability of the i^{th} element in a system composed of N elements. For the case where the system elements are redundant,

$$R_o = 1 - (1 - R_1)^2 \times 1 - (1 - R_2)^2 \times 1 - (1 - R_n)^2.$$

2.6.3.2 Compilation of Failure Rate Data

Historical failure data were collected from both Boeing and external sources. To the extent possible, the data were classified into three general categories:

a) Existing equipments,

b) Equipments under development, and

c) Proposed equipments.

In addition, these data were identified according to: (1) source, i.e., company or programs, and (2) type, i.e., predicted or achieved. Appendix A-4 presents these data in detail.

2.6.3.3 Reliability Feasibility Analysis

Table 2.6-2 presents a capsule summary of the inherent reliability feasibility ranges for the major subsystems in navigation and guidance systems. These data are derived from the historical data presented in Appendix A-4 and are used to compute the system reliabilities in the preceding section. In addition, the data in Table 2.6-2 are based on the following conditions:

a) The subsystems are developed or proposed to be developed by the end of 1968,

b) The operating environment is benign, and

c) The lower end of the feasibility ranges is obtained when standard military parts

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are used and the higher end of ranges require the use of Minuteman type parts.

Figure 2.6-1 illustrates the reliability improvement gained by the Minuteman program over a military program that used MIL-Spec parts. The Minuteman part failure rates are based on over 5,000,000 system operating hours in silo environment, while the MIL-Spec part failure rates were derived from nearly a million system hours of Navy operational vehicles.

Table 2.6-3 presents typical mission data used to compute the system reliability feasibility ranges.

2.6.3.4 Other Factors Affecting Reliability

The hardware development time period for this study is assumed to be 1975 or later. Because this period is at least 7 years beyond 1968, the date for which the reliability feasibility ranges were assumed valid, it is anticipated that the upper end of the reliability feasibility ranges will be higher. Just how much higher is difficult to establish at this time due to inadequate data. However, there are many factors that are expected to have a significant influence on this upward trend. One of the factors that warrant consideration during this study is the adjustment or allowance for the environmental severity of the flight.

Experience shows that it is usually necessary to employ adjustment factors (K) when reliability assessments of a system are made using data from systems having dissimilar environments. In this study, adjustment for flight environment is considered to be paramount. Since the failure data were from a benign environment, this adjustment is expected to lower or degrade the system's reliability.

How much degradation to expect from this adjustment was given considerable attention. Table 2.6-4(1) is a compilation of some currently used environmental K-factors. For some environments these K-factors are in good agreement, for others there is a considerable difference. In this study a K-factor of 5 was used for all phases of flight except the second stage thrust and coast phases, where 125 and 1 were used, respectively. These values represent average values of the K-factors developed by Boeing for the supersonic transport (SST) design.(2)

Although it is usually difficult to establish functional relationship between cost and reliability, cost of parts is expected to have an impact on the system's reliability.

Initial parts cost is affected by a number of elements such as time as related to product life cycle, cost of manufacture, competition, and existing market situations. For parts having rapid technological growth these factors combine to produce an average selling price that can vary extensively during its product life cycle.

( ) denotes reference

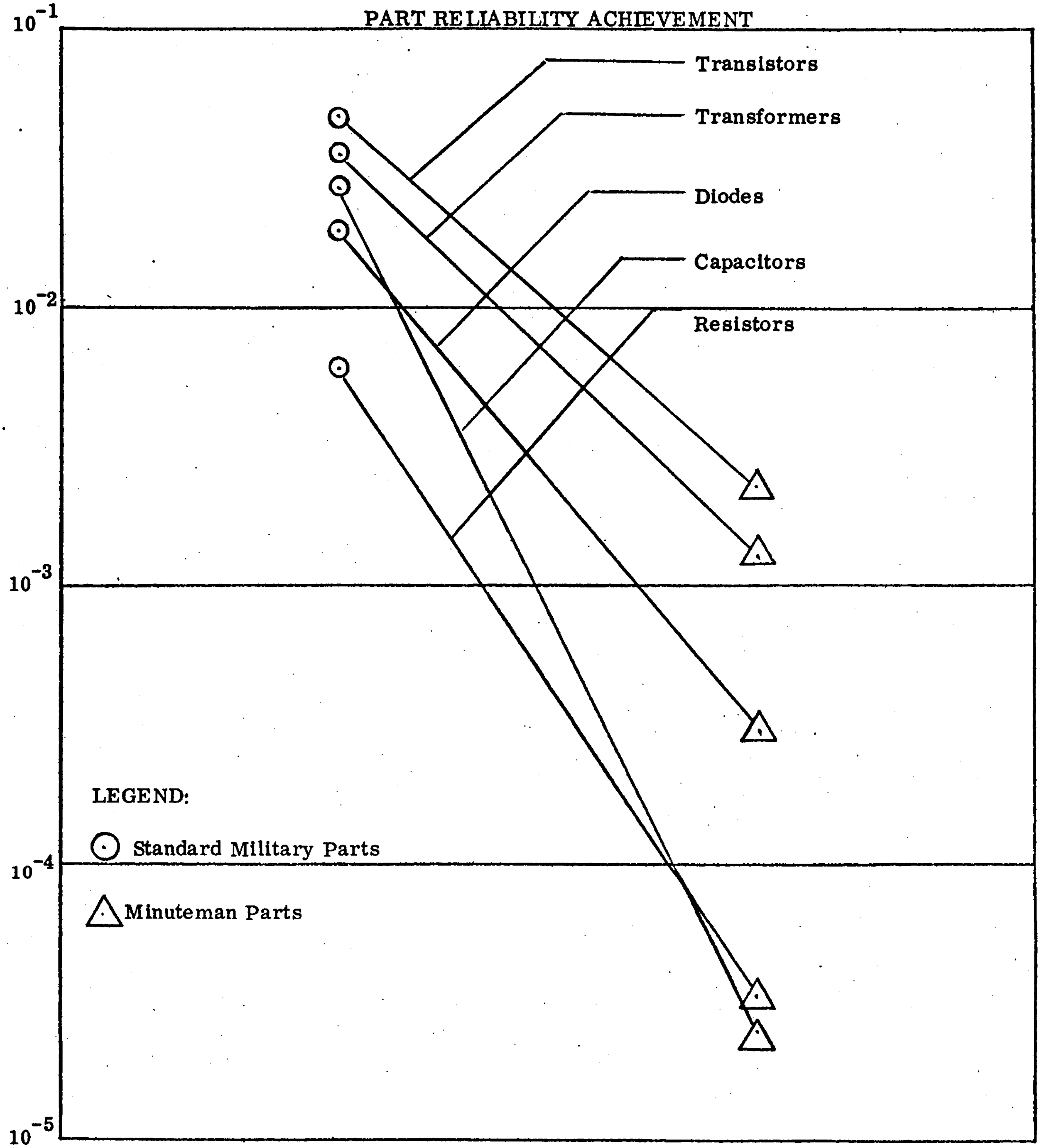
PART RELIABILITY ACHIEVEMENT
Transistors
Transformers
Diodes
Capacitors
Resistors
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
Failure Rate
LEGEND:
Standard Military Parts
Minuteman Parts

FIGURE 2.6-1

A COMPILATION OF SOME CURRENTLY USED K_e FACTORS

TABLE 2.6-4

Environment	Earles & Eddins	MIL-STD-756A	Ryerson-* Hughes Part Type Failure Rates	Space Aeronautics Volume 42, No. 5, Oct. 1964, page 53	Farada Source 93	D2-100255 Boeing Lunar Orbiter	D2-23095 Boeing Failure Modes & Effects Analysis	D2-23834-1 Voyager 71 Reliability Analysis & Prediction
Laboratory Computer	1		1	1			1	1
Satellite (in orbit)	1	1			1	1	0.9	1
Manned Spacecraft (in orbit)							1.5	
Fixed Ground Equipment	8	1	1.05 - 1.67	10	1	1	2	3 - 5
Shipboard	15	1			2		3	
Rail Mounted	22						3.5	
Aircraft (in flight)	50	6.5	1.4 - 167	150	2		6	
Missile Equipment (in flight)	900		2.75 - 13,400	1000	8		91	150 - 1200
Missile Nose Cone (in flight)	800	80				140	82	
Maneuvering in Space								3 - 20

A COMPILATION OF SOME CURRENTLY USED K_e FACTORS - Continued

TABLE 2.6-4 (Continued)

Environment	Earles & Eddins	MIL-STD-756A	Ryerson-* Hughes Part Type Failure Rates	Space Aeronautics Volume 42, No. 5, Oct. 1964, page 53	Farada Source 98	D2-100255 Boeing Lunar Orbiter	D2-23095 Boeing Failure Modes & Effects Analysis	D2-23834-1 Voyager 71 Reliability Analysis & Prediction
Between Tanks Stage II (in flight)	900						92	
Sustainer Engine Compartment (in flight)	1000						100	
Between Tanks Stage I (in flight)	1250						125	
Booster Engine Compartment	1500						150	
Trailer Mounted Support Equipment	20				1.5			

* Ryerson uses fixed ground environment as a base and computes K_e factors for the Laboratory Computer Environment at some value less than 1. For comparison purposes, these were normalized to 1 and all other values adjusted accordingly.

(Continued)

Figure 2.6-1 illustrates quite vividly how the failure rates of some parts, used on current military programs vary as a function of procurement policy, i.e., whether the part is of the standard military variety that requires a nominal amount of testing, quality control, etc., or it is of the hi-rel variety which requires considerably more testing, screening, etc.

A study by Rand Corporation indicates that the reliability of integrated circuits is also quite sensitive to the type procurement policy used.(3)

For example, the study shows that an integrated circuit with a complexity equivalent to 30 discrete parts has a failure rate of 116 percent per 1000 hours when procured as a standard military grade compared to a failure rate of 29 percent per 1000 hours when procured as a high reliability grade. The predicted prices (dollars) for this integrated circuit in quantities of 10,000 were as follows:

Year	1965	1968	1970
Standard Military	20	10	4
Hi-Rel	30	17	9

In conclusion, the results of this phase of the study indicate that from the standpoint of reliability and part cost the design reliability objective of 0.99 and a mission reliability goal of 0.95 can be met in an economical manner.

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References

1. RSS 65-11, "Failure Frequency Adjustment Factors (K_e) for Various Environmental Conditions," The Boeing Company, October, 1965.

2. D6A10095-1, "Reliability and Maintainability Prediction Standard - SST Program," The Boeing Company, September 3, 1966.

3. RM-4511-ARPA, "The Reliability of Ground-Based Digital Computers," The Rand Corporation, June, 1965.

2.7 COST ANALYSES

A cost model has been developed for comparing alternate navigation and guidance concepts. An effort has been made to average cost data from various sources and to explain trends to be expected. Factors that contribute to cost differences include (1) the development effort required to meet program requirements, (2) differences in functional capability, (3) differences in performance, for example, accuracy, (4) the extent of the reliability program and testing effort, (5) differences in equipment complexity, (6) differences in technology, for example, discrete electronic parts versus integrated circuits. It is difficult to attribute differences in the available cost summary data to these various potential influencing factors.

Program costs are divided into non-recurring and recurring categories. The non-recurring costs are a measure of the research, design, development, and test effort required to obtain the first flight article. Recurring costs are the costs of obtaining additional units on a production line manufacturing and test basis. The comparative cost data developed thus far has been primarily for recurring costs.

Recurring costs depend on the number of units produced on a learning curve. The learning curve experience has varied between 85% to 95% for different programs. A 90% learning curve has a second unit cost 90% of the second, and a 200th unit cost 90% of the 100th unit. Figure 2.7-1 shows how the unit cost varies with the number of the unit produced for various learning curves. The initial unit recurring cost is a means of comparing alternate system designs if they can be produced on the same learning curve. If different learning curves exist then the comparison should be on the basis of relative total recurring cost for the number of units required for the program under consideration.

Two types of inputs have been used in developing a cost model for alternate navigation and guidance systems. The first approach is to build up the cost of a system from component costs. The second approach is to summarize industry cost proposals for overall systems. The two approaches should give the same answer for a consistent model.

Cost estimates for a number of basic navigation sensor types are given in Table 27-1. The cost of gyro and accelerometer inertial components are a function of their accuracy performance. The one sigma random drift rate has been used as a figure of merit for labeling gyro cost classes; and bias error has been used to characterize accelerometer performance. The cost performance correlation is based on experience in testing inertial components in the Boeing inertial laboratory, on the cost for the purchase of a number of different components, and on industry component cost quotations. The cost estimates for doppler radars, star trackers, astrocompass, and vertical references have been based on several typical values for each component. Three points on a cost learning curve have been estimated to help explain differences in quotations. The learning curve used

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is a typical one (91%) for overall navigation systems. The astrocompass costs are greater than the star tracker estimates because of the inclusion of the astrocompass computation functions in current designs.

The inertial platform and associated platform electronics cost estimates given in Table 2.7-2 have been based on both types of cost inputs. The first unit costs can be obtained to a good approximation by taking the gyro and accelerometer costs for the number of components used in a platform and multiplying by a factor of two to obtain the platform plus electronics costs, the cost estimates for production runs are averages of industry cost proposals. A 91% cost learning curve has been used to relate production costs to first unit costs. A close correlation has been obtained between the two ways of estimating the costs. The platforms are assumed to have three gyros and three accelerometers except for the 0.001 degree/hour platform which has two two-degree-of-freedom gyros. Higher performance accelerometers have been used for the Stage 2 inertial platforms since Stage 2 experiences much higher accelerations than Stage 1 with a resulting greater error sensitivity.

Computer capability and cost trade data are given in Table 2.7-3. These data were obtained from another Boeing preliminary design effort and have included normalized relative cost estimates for a family of preliminary design computer configurations made by Autonetics. Dollar estimates for several preliminary designs were made by Boeing by developing component costs and construction costs into an estimate of the overall computer cost by standard cost estimating procedures used for cost proposals. Memory unit costs were used to relate the dollar estimates for the several Boeing designs to the normalized relative cost data. The resulting computer cost estimates then checked very closely with several independent cost estimates by two other computer manufacturers.

The computer cost estimates have been made for configurations with only a basic minimum capability. The data words and instruction words have 16 bits per word. There are 16 instructions in the computer control. Destructive readout (DRO) memory has been used. A minimum required input-output capability has been used. The costs are based on the use of microelectronic integrated circuits with a significant cost reduction compared to designs of several years ago using discrete electronic parts. The effect of these assumptions is to make the computer costs less than the cost of typical current navigation and guidance computers. However, since computer costs for a given capability are expected to go down with time the Table 2.7-3 estimates are believed realistic for this trade study.

The costs of navigation and guidance systems are estimated by combining sensor costs from Table 2.7-2 with computer costs from Table 2.7-3. Some possible combinations for Stage 1 navigation systems are given in Table 2.7-4, and Stage 2 possibilities are given in Table 2.7-5. In this application with a Mach 7 cruise speed for Stage 1 the doppler radar and astrocompass systems are less accurate and more expensive than a low accuracy, 37 km/hr. (20 N.M./hr), inertial navigator. On this basis these systems (1 and 2 of Table 2.7-4) can be eliminated from further consideration. Table 2.7-6 shows the range of costs of the launch vehicle guidance and navigation system using different combinations of inertial systems in the Stage 1 and Stage 2 vehicles.

FIGURE 2.7-1 COST LEARNING CURVE

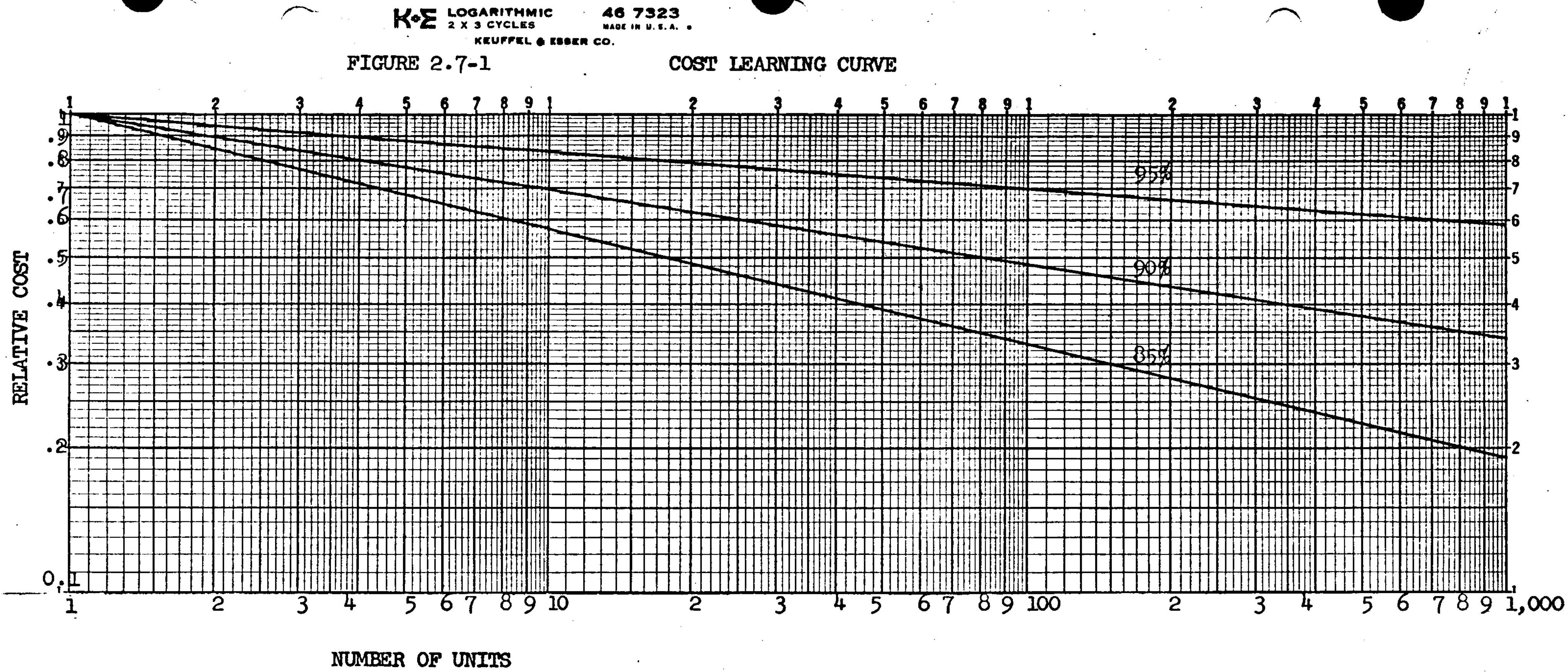

TABLE 2.7-1 NAVIGATION SENSOR COMPONENT COST ESTIMATES

Component	Characteristic	Cost in Thousands of Dollars: 1st Unit	Average of 50 Units	Average of 300 Units
Gyros	Average random drift rate in degrees/hour			
	1.	7		
	0.1	14		
	0.01	20		
	0.001	45.		
Accelerometers	Bias error (g)			
	10^{-4}	2		
	10^{-5}	20.		
	10^{-6}	40.		
Doppler radar		40.	28	22.
Star Tracker		37.	26	20.
Astrocompass		74.	51.	40.
Vertical reference	Gyro drift (degree/hour)			
	0.5	25	17	12.
	0.1	84	60	47.

TABLE 2.72 INERTIAL PLATFORM AND PLATFORM ELECTRONICS COSTS

(91% cost learning curve assumed)

Performance, CEP Navigation Accuracy	Gyro	Accelerometer	Cost (In Thousands of Dollars) 1st Unit	Average of 50 Units	Average of 300 Units
Stage 1					
20 NM/hr	0.1 deg/hr	10^{-4} g	82.	57.	44.
2 NM/hr	0.02	5×10^{-5}	144.	99.	78.
1 NM/hr	0.01	3×10^{-5}	180.	124.	97.
0.5 NM/hr	0.005	3×10^{-5}	217.	150.	117.
0.1 NM/hr	0.001	10^{-5}	300.	207.	162.
Stage 2					
	1. deg/hr	10^{-4}	54.	38	30.
	0.1	10^{-4}	82.	57.	45.
	0.01	10^{-5}	240.	165.	130.
	0.001	10^{-6}	410.	282	221.

TABLE 2.7-3 COMPUTER CAPABILITY - COST TRADE DATA

16 bits/word

	Capability			Cost (In Thousands of Dollars)		
		Computation Speed (in microseconds)				
Type	Memory Size	Add	Multiply	1st Unit	Average of 50	Average of 300
Analog, navigation				18	13.	10.
DDA	87 integrators			32	20	17.
GP, series	512	20	260	33	20.3	17.5
GP, series	4,096	20	260	60.	45.	36.
GP, series	8,192	20	260	76.	60.	48.
GP, series	16,384	20	260	109.	90.	72.
GP, parallel	4,096	6	40	162.	124.	101.
GP, parallel	8,192	6	40	188.	143.	129.
GP, parallel	16,384	6	40	213.	163.	144.

Table 2.7-4 STAGE 1 NAVIGATION SYSTEM COST ESTIMATES

Sensors	Computer	Accuracy	Cost (In Thousands of Dollars) 1st Unit	Average of 50	Average of 300
1. Doppler radar, astrocompass, 0.5°/hr Vertical	Analog	1%	157.	109.	84.
2. Doppler radar, astrocompass 0.1°/hr Vertical	4K word, GP digital, series	0.2%	258	184	145
3. Inertial	"	20NM/hr, 37 km/hr.	142.	102.	80.
4. Inertial	"	20NM/hr, 37	204	144	114.
5. Inertial	"	1NM/hr, 1.85	240.	169	133
6. Inertial	"	0.5NM/hr, 0.62	277.	195.	153
7. Inertial	"	0.1NM/hr, 0.18	360	252.	198.
8. Doppler Inertial 0.1 deg/hr gyros	"		182	130.	102
9. Doppler Inertial 0.01 deg/hr gyros	"		244	172.	136
10. Stellar Inertial, 0.1 deg/hr gyros	"		179.	128.	100.
11. Stellar Inertial, 0.01 deg/hr gyros	"		241.	170	134.
12. Stellar Doppler Inertial, 0.1 deg/hr gyros	"		219.	156.	122.
13. Stellar Doppler Inertial, 0.01 deg/hr gyros	"		281.	198.	156.

Table 2.7-5 STAGE 2 NAVIGATION AND GUIDANCE SYSTEM COST ESTIMATES

Sensors	Computer	Accuracy	Cost (In Thousands of Dollars) 1st Unit	Average of 50	Average of 300
1. Inertial	Minimum	1°/hr gyro	87	58.	48.
2. Inertial	4K word, GP digital, series	0.1°/hr	142.	102.	81.
3. Inertial	"	0.01°/hr	300.	210.	166.
4. Inertial	"	0.001°/hr	470	327.	257.
5. Stellar Inertial	"	1°/hr gyro	124	84.	68.
6. Stellar Inertial	"	0.1°/hr	179	128	101.
7. Stellar Inertial	"	0.01°/hr	333.	236	186.

Table 2.7-6

LAUNCH VEHICLE GUIDANCE & NAVIGATION COSTS

Stage 1		Stage 2	Total Cost
20 NM/hr	37 km/hr.	1°/hr, Min Computer	$202K
1 NM/hr	1.85	1°/hr	$327K
0.1 NM/hr	0.18	1°/hr	$447K
20 NM/hr	37	0.1°/hr	$257.K
1 NM/hr	1.85	0.1°/hr	$382.K
0.1 NM/hr	0.18	0.1°/hr	$502.K
20 NM/hr	37	0.01°/hr	$415K
1 NM/hr	1.85	0.01°/hr	$540K
0.1 NM/hr	0.18	0.01°/hr	$660K
20 NM/hr	37	0.001°/hr	$585.K
1 NM/hr	1.85	0.001°/hr	$710.K
0.1 NM/hr	0.18	0.001°/hr	$830.K

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2.8 Tradeoffs

Performance, reliability and cost estimates are combined in the trade off comparison of alternate navigation and guidance concepts. The proposed approach to a measure of the overall performance of a navigation and guidance system is based on payload weight. The assumption is made that the payload effectiveness is proportional to its weight. This is reasonable since payload maneuvering capability is approximately proportional to fuel weight, electrical subsystem capability is proportional to its weight, mission time capability depends on the weight of expendables, and the number of sensors that the payload can carry is determined by their weight.

A reference payload weight W_p = 6,200 kg (13,720 lbs.) is defined for an ideal navigation and guidance system - one that has zero weight, zero errors, and an ideal guidance law with zero fuel penalty. The payload weight penalty due to navigation and guidance is then determined for each concept under consideration. This penalty is the sum of the navigation and guidance equipment weights, the fuel penalty due to a non-ideal guidance law, and the fuel penalty to correct errors. The second stage weights subtract directly from the payload. The Stage 1 navigation and guidance effects on the payload weight is scaled down by the ratio:

$$\frac{\text{Stage 2 weight after rendezvous}}{\text{Initial stage 2 weight}} = 0.109$$

The Stage 1 penalties include the fuel weight to correct the return phase errors to reach the landing field given in Section 2.5.4.

The total payload weight penalty is ΔW due to both the first and second stage navigation and guidance systems. The relative performance effectiveness is

$$\frac{W_p - \Delta W}{W_p} \tag{1}$$

where W_p is the reference payload weight. The navigation and guidance reliability for the specified mission is designated R. Then, the overall relative effectiveness is

$$\frac{W_p - \Delta W}{W_p} \times R \tag{2}$$

The total payload weight penalties have been obtained by adding the rendezvous fuel weight, (2.5.2) Stage 1 equipment weight modified by the exchange ratio, Stage 2 equipment weight, (2.3.3), and the rendezvous radar weight for the terminal phase three sigma position error (Appendix A3). The resulting total payload weight penalty for the rendezvous mission (return phase is not included) is given for selected navigation and guidance systems in Figure 2.8-1.

The total vector velocity error at Stage 2 thrust cutoff for injection into the rendezvous transfer orbit has been used to characterize the accuracy performance for the system concepts given in Section 2.3.3. The total payload weight penalty is plotted versus this total velocity error in Figure 2.8-2. This figure can be used to estimate the total weight penalty for system configurations not summarized in Figure 2.8-1. The relative performance effectiveness, from Equation (1), is given versus the total velocity error at transfer injection in Figure 2.8-3.

The overall relative effectiveness - weight performance X reliability - is plotted as a function of guidance accuracy in Figure 2.8-4. High and low values of estimated reliability from Table 2.6-1 have been used. The system reliability is the product of the Stage 1 and Stage 2 navigation and guidance system reliabilities. Also the Figure 2.8-3 relative performance trade has been modified to include the Stage 1 return-to-base navigation error-payload trade from Section 2.5.4. The results are given in Figure 2.8-4 as six curves: high and low reliability estimates, and 18.5, 1.85, and 0.18 km/hr. (10, 1, and 0.1 N.M./hour) Stage 1 navigation accuracies. There is very little difference in the effectiveness of the 1.85 km/hr (1 N.M./hr. and 0.18 km/hr. (0.1 N.M./hr) systems for the low reliability valves, and the curves are superimposed for the high reliability valves. This indicates that for an effectiveness - accuracy trade there is no significant advantage to a 0.18 km/hr. (0.1 N.M./hour) Stage 1 system.

Recurring cost data from Section 2.7 is plotted versus the total vector velocity characteristic error in Figure 2.8-5 for the three Stage 1 navigator accuracy classes. Overall effectiveness from Figure 2.8-4 and cost from Figure 2.8-5 are cross plotted on Figure 2.8-6 to give a cost - effectiveness trade. The low and medium accuracy Stage 2 systems (1 degree/hour and 0.1 degree/hour gyros) can be eliminated from this trade. Also, the very high accuracy Stage 2 system, 0.001 degree/hour gyro class, gives a very marginal increase in effectiveness for a large increase in cost; and thus, can be eliminated. This narrows the consideration to the 0.01 degree/hour stage 2 accuracy class; but does not distinguish between the 18.5, 1.85, and 0.185 km/hr (10, 1, and 0.1 N.M./hour) Stage 1 navigators. An 0.5% increase in effectiveness is obtained for about $125,000 increase in recurring cost in going from 18.5 km/hr. (10 N.M./hour) to 1.85 km/hr. (1 N.M./hour), or from 1.85 km/hr. (1 N.M./hour) to 0.185 km/hr. (0.1 N.M./hour). The selection of the Stage 1 system has been established by the qualitative factors discussed in Section 2.9.

Aided inertial systems - doppler inertial and stellar inertial - have not been considered in this effectiveness trade because they create development problems and installation problems that are expected to significantly increase the non-recurring development costs.

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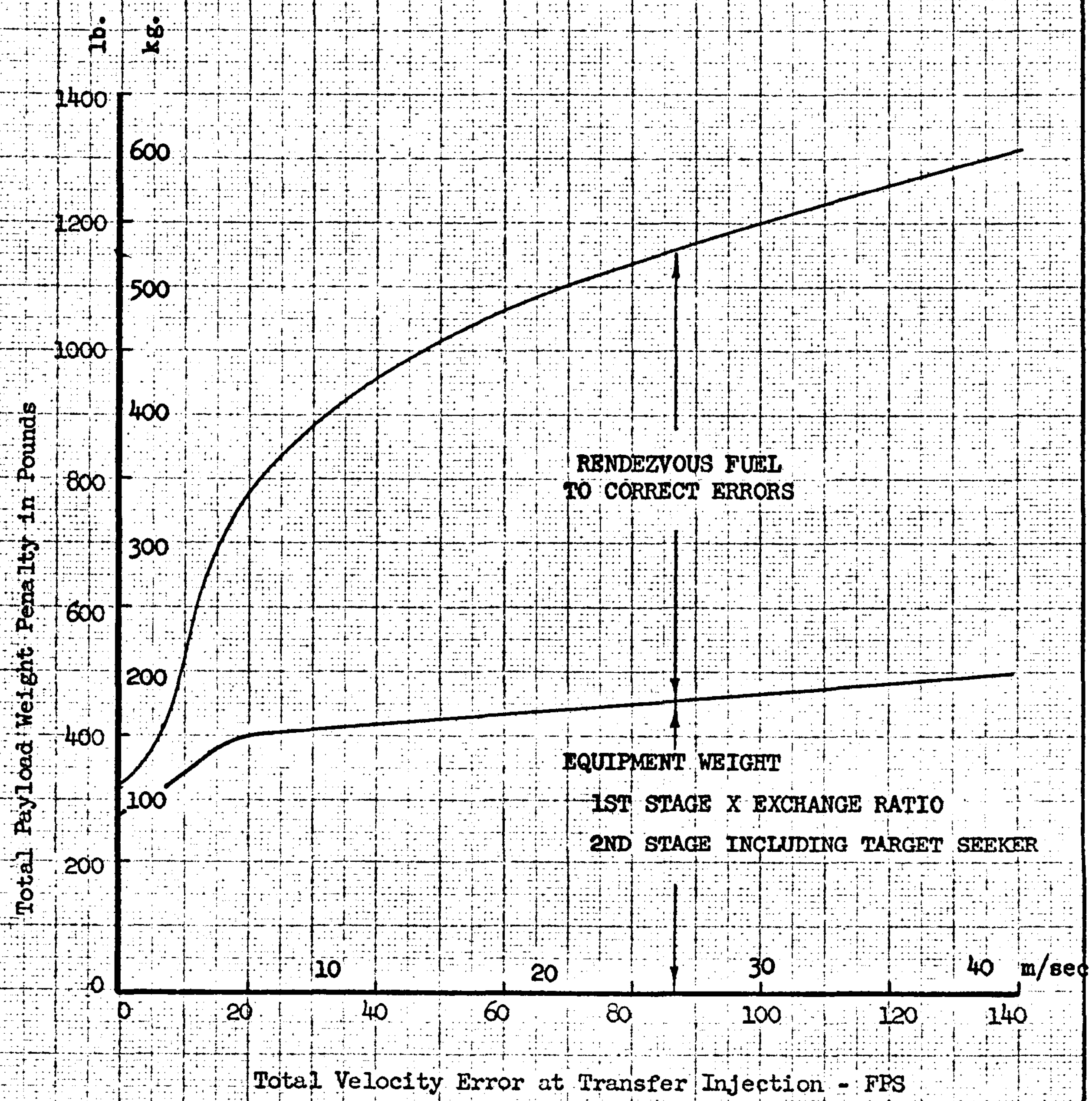
FIGURE 2.8-1 TOTAL PAYLOAD WEIGHT PENALTIES FOR GUIDANCE
RENDEZVOUS MISSION, 20 MINUTE TERMINAL MANEUVER CLOSING TIME

	System Classification				Payload Weight Penalties									
	Guidance Capability		Navigation Accuracy		Rendezvous/Equipment-Wt.						Rendezvous Radar Wt.		Total	
					Fuel		Stage 1		Stage 2					
No.	Stage 1	Stage 2	Stage 1	Stage 2	kg	(lbs)	kg	(lbs)	kg	(lbs.)	kg	(lbs)	kg	(lbs.)
4	Max	Med	Med	Low	306	(675)	16	(35)	92	(202)	91	(200)	504	(1110)
6	Max	Max	Med	Med	154	(340)	16	(35)	99	(218)	61	(135)	330	(728)
7	Max	Max	Med	Med+	122	(270)	16	(35)	99	(218)	57	(125)	294	(648)
10	Max	Max	Med	High	36	(80)	16	(35)	99	(218)	36	(80)	187	(413)
13	Max	Max	Med	High+	20	(45)	16	(35)	99	(218)	32	(70)	167	(368)



	INITIALS	DATE	REV BY INITIAL	DATE	TITLE	MODEL
CALC					Performance Weight Penalties vs. Guidance Accuracy	FIGURE 2.8-2
CHECK						
APPD.						
APPD.					Rendezvous Only	

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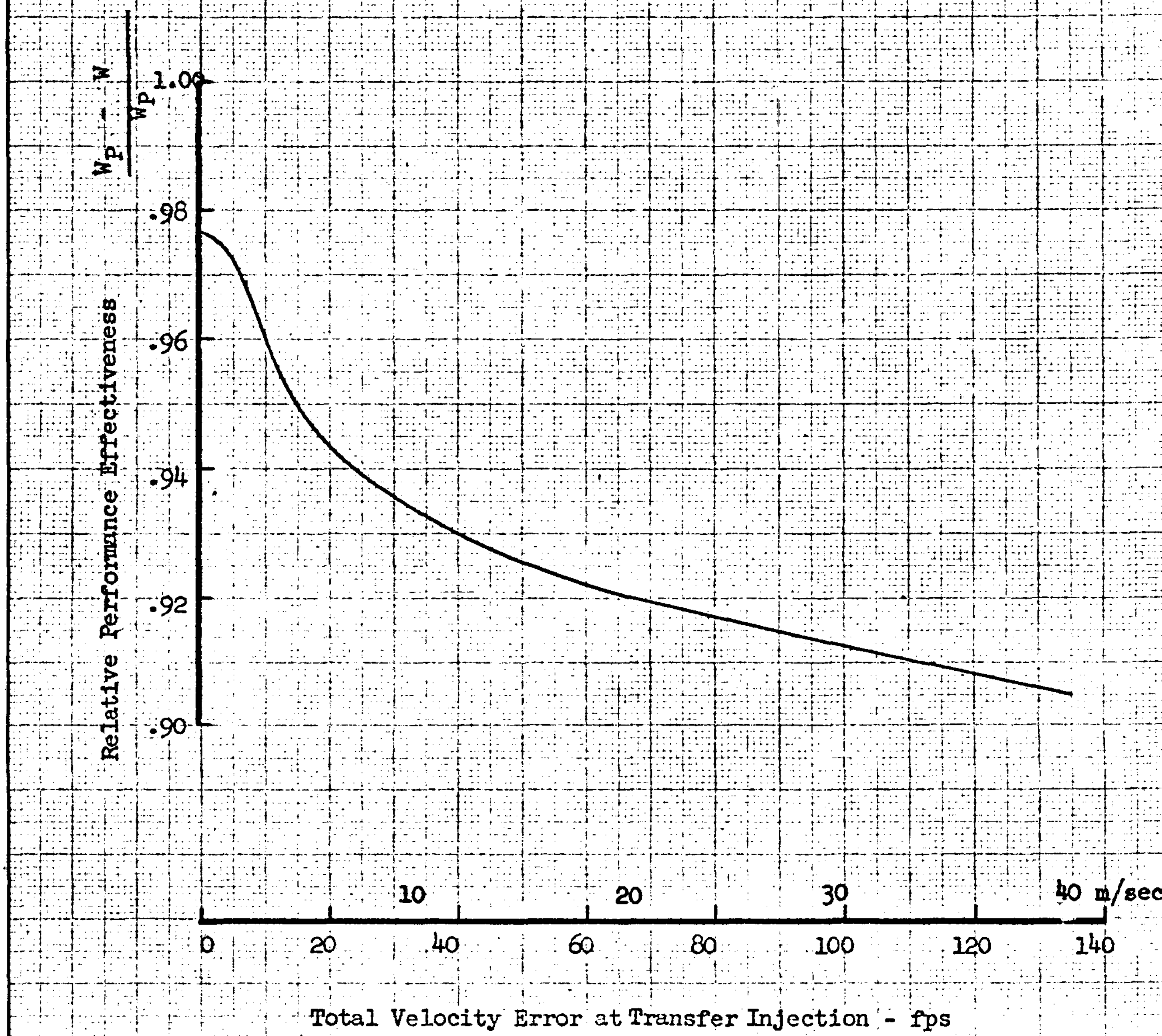
REV LTR



	INITIALS	DATE	REV BY INITIAL	DATE	TITLE	MODEL
CALC					RELATIVE PERFORMANCE EFFECTIVENESS VS. GUIDANCE ACCURACY	FIGURE 2.8-3
CHECK						
APPD.						
APPD.						

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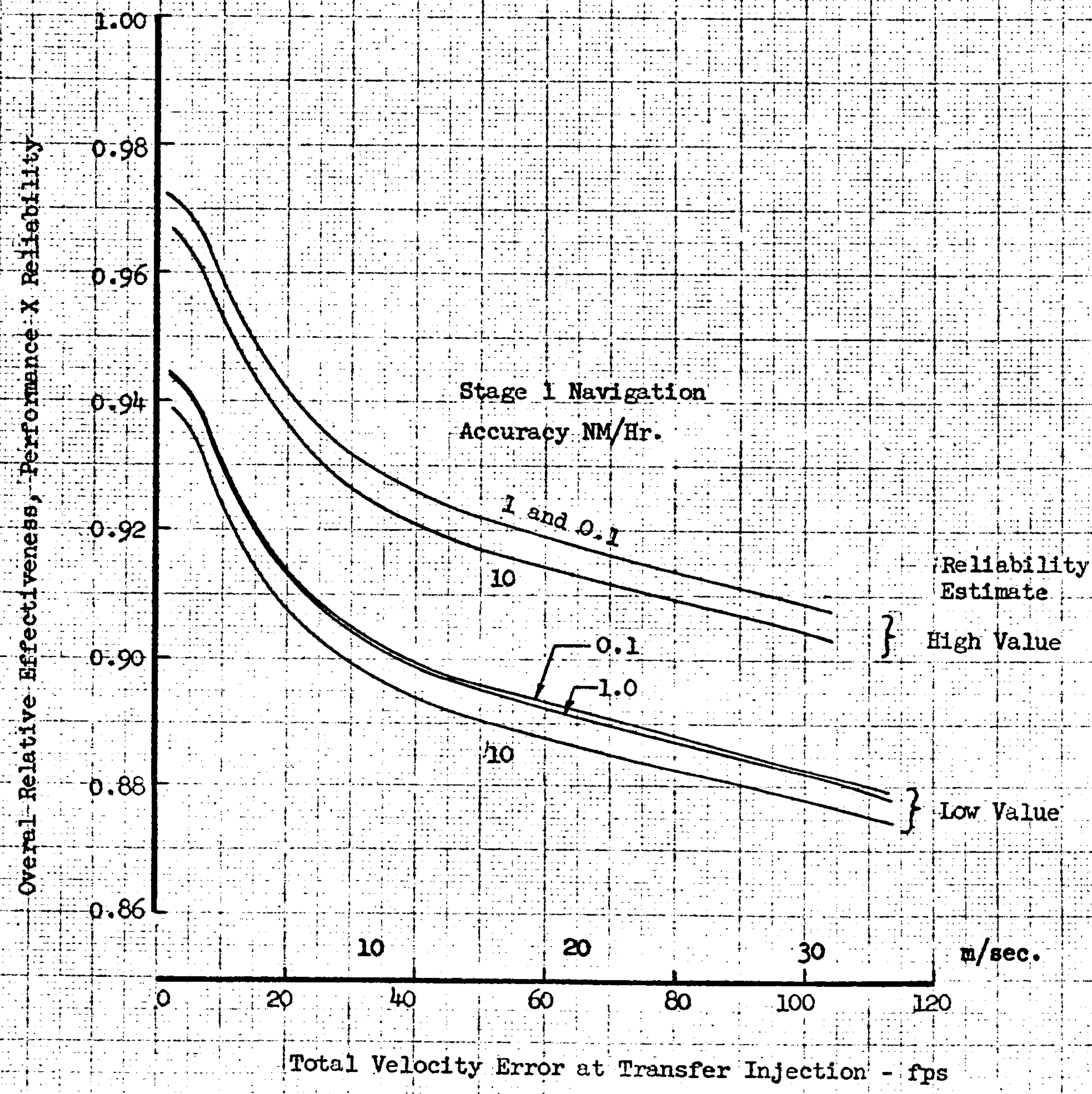

	INITIALS	DATE	REV BY INITIAL	DATE	TITLE	MODEL
CALC					OVERALL RELATIVE EFFECTIVENESS VS. GUIDANCE ACCURACY	FIGURE 2.8-4
CHECK						
APPD.						
APPD.						

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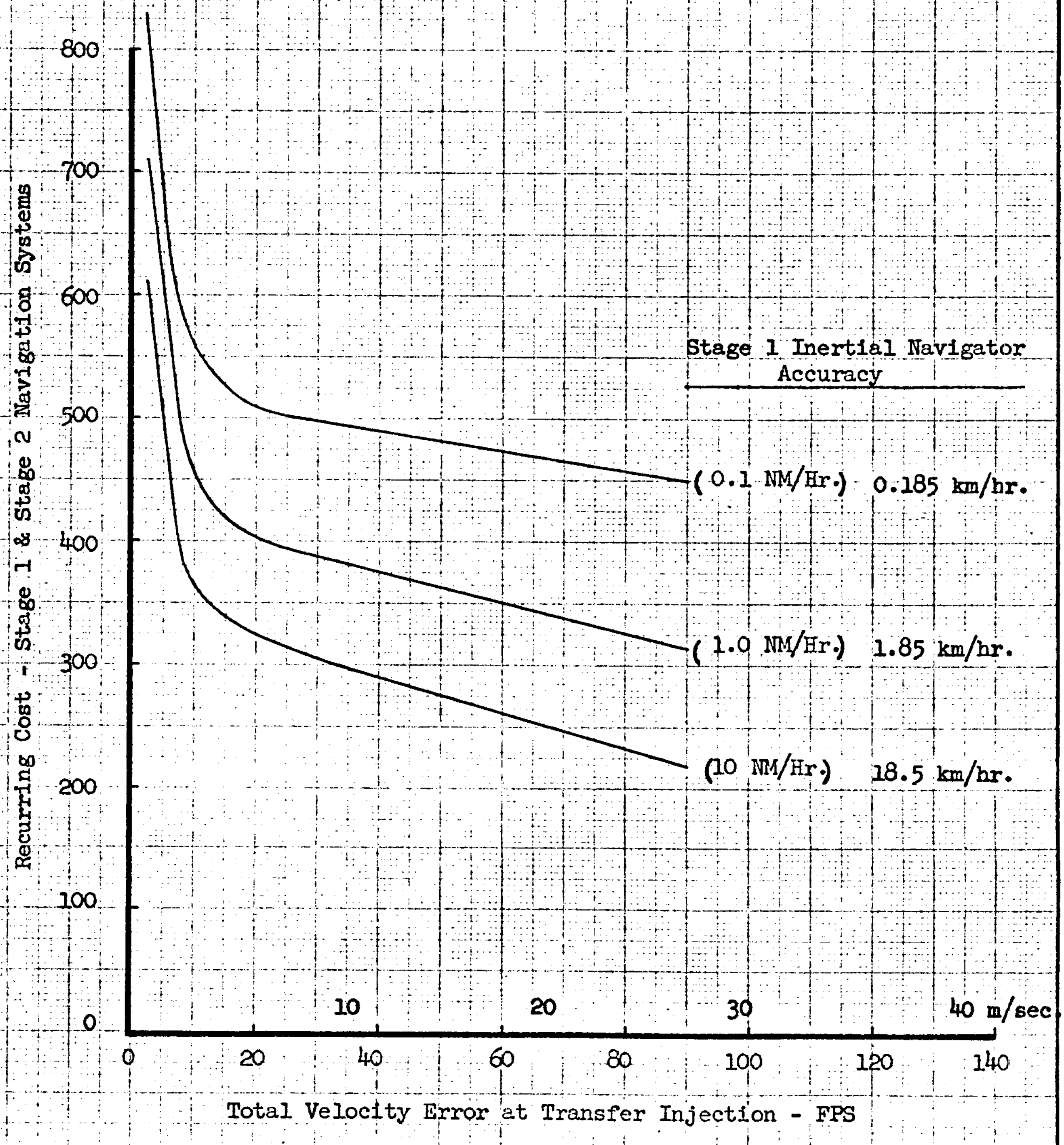
REV LTR



	INITIALS	DATE	REV BY INITIAL	DATE	TITLE	MODEL
CALC					COST VS. GUIDANCE ACCURACY	FIGURE 2.8-5
CHECK						
APPD.						
APPD.						

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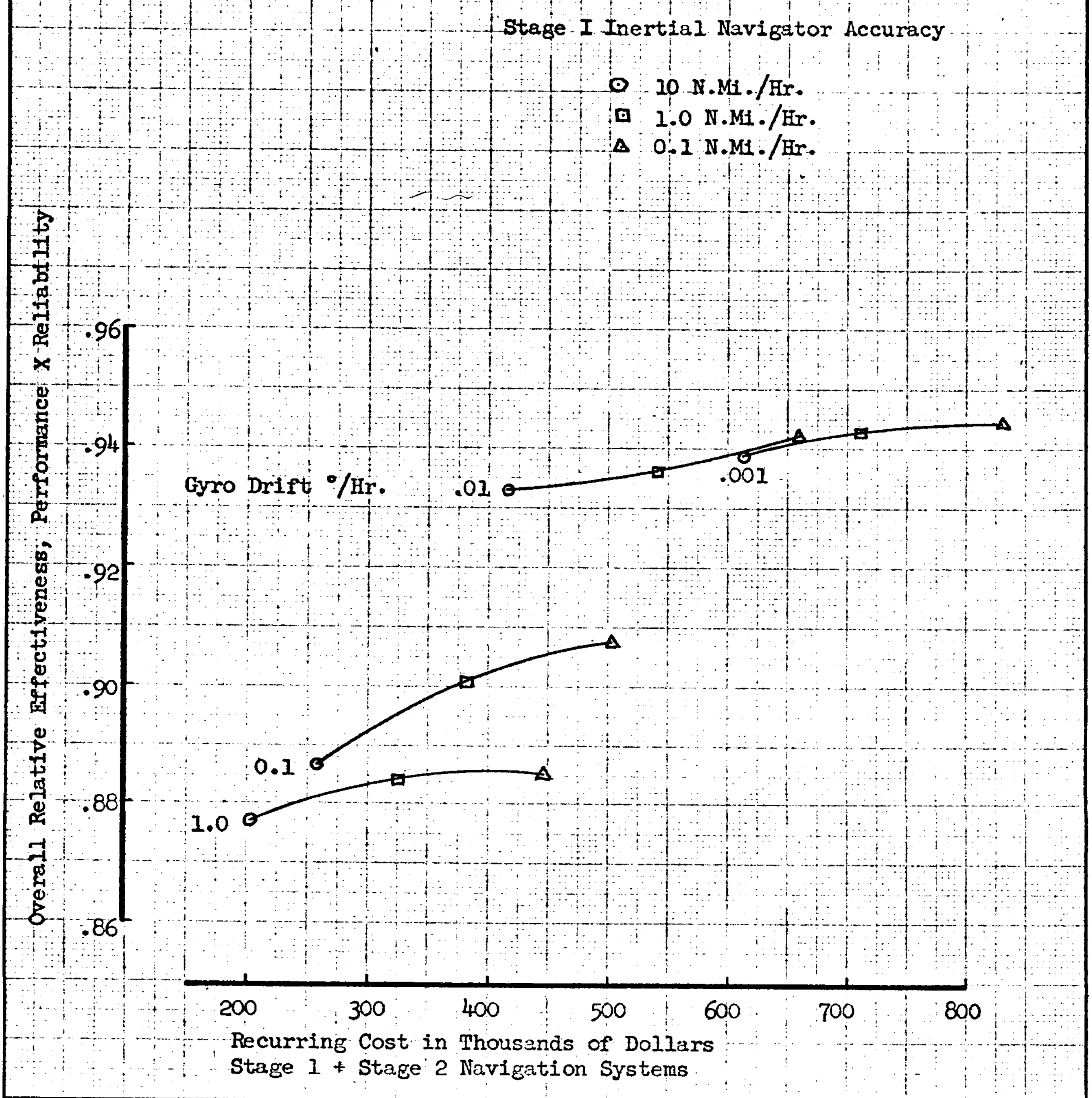
REV LTR



	INITIALS	DATE	REV BY INITIAL	DATE	TITLE	MODEL
CALC					EFFECTIVENESS VS. COST	FIGURE 2.8-6
CHECK						
APPD.						
APPD.						

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2.9 RECOMMENDED NAVIGATION AND GUIDANCE CONCEPTS

Two navigation and guidance concepts are recommended for detailed Phase 2 study. One of these concepts has been selected from current or near term technology with the objective of minimizing development costs and also minimizing performance penalties. The second concept has been selected from potential or advanced concepts to determine the payoff for additional technology development effort to the launch vehicle capabilities.

2.9.1 RECOMMENDED CURRENT TECHNOLOGY CONCEPT

Stage 1 Navigation and Guidance

(1) Pure inertial platform basic navigation technique, 1.85 km/hr. (1NM/Hr) CEP accuracy class.
(2) Flexible general purpose digital computer for implementing multiple mode guidance methods.
(3) Air data system - air density and true air speed - to minimize performance penalties associated with flight profile constraints and to provide needed data to the pilot for takeoff and landing.
(4) Radar altimeter (or pressure altimeter) for stabilizing the inertial navigator vertical channel.
(5) Pilot displays and controls for operation of the system and monitoring performance.
(6) VOR, Tacan, and ILS (instrument landing system) radio navigation aids for descent and landing field approach under all weather conditions.
(7) Refueling rendezvous radio aid (study required to determine if additional aid is needed for this function).
(8) Conventional communications receiver for target satellite orbit updating.
(9) Redundancy to meet reliability and flight safety requirements.

Stage 2 Navigation and Guidance - Manned Flight

(1) Pure inertial platform - 0.01 degree/hour gyro drift rate and 3×10^{-5}g accelerometer bias accuracy class, preflight level and alignment.
(2) Flexible general purpose digital computer for implementing near minimum fuel guidance law, direct ascent or parking orbit options, rendezvous terminal phase guidance, in-orbit navigation and guidance if required, and deorbit and reentry navigation and guidance.
(3) Terminal phase acquisition and tracking radar for uncooperative targets, with a cooperative target mode.
(4) Pilot displays and controls for operation of the system and monitoring performance.
(5) Communications for ground assistance in emergencies.
(6) Redundancy to meet reliability and flight safety requirements.

Stage 2 Navigation and Guidance - Unmanned Flight

(1) Same inertial platform.
(2) Simplified digital computer.
(3) Same acquisition and tracking radar.
(4) No displays or controls.
(5) Communications link for mode control and override.
(6) No redundancy.

2.9.2 RECOMMENDED ADVANCED TECHNOLOGY CONCEPT

Stage 1 Navigation and Guidance

(1) Strapdown inertial system, 1.85 km/hr (1 NM/hour) CEP accuracy class.
(2) Flexible general purpose digital computer, statistical data processing for navigation accuracy improvement using auxiliary sensor data, lambda matrix guidance law.
(3) Omega and navigation satellite position updating.
(4) Air data system and altimeter.
(5) VOR, Tacan, ILS descent and landing radio aids.
(6) Refueling rendezvous radio aid (if needed).
(7) Conventional communications receiver.

Stage 2 Navigation and Guidance - Manned Flight

(1) Strapdown inertial system.
(2) Flexible general purpose digital computer, DDA for coordinate transformations, lambda matrix guidance law.
(3) Terminal phase acquisition and tracking radar.
(4) Pilot displays and controls.
(5) Communications link.

Stage 2 Navigation and Guidance - Unmanned Flight

(1) Same strapdown inertial system.
(2) Simplified digital computer.
(3) Same acquisition and tracking radar.
(4) Communications link.

2.9.3 Discussion of the Recommendations

A 1.85 km/hr. (1 N.M./hour) CEP accuracy class Stage 1 system has been selected for several reasons. Flight safety is increased because air traffic control is much more effective and the probability of air collision is reduced. This is particularly important in the approach to the airfield when navigation errors are large. The refueling rendezvous task is also easier with accurate navigation. An inertial navigator has been selected. Considerably more development effort has gone into the 1.85 km/hr. (1 N.M./hour) class system than either 18.5 km/hr. (10 N.M./hour)or 0.185 km/hr. (0.1 N.M./hour) system. Development cost for a 1.85 km/hr. (1 N.M./hour) system will not be excessive.

The selection of inertial navigation systems follows naturally from the mission profile and mission constraints. The short mission time, high altitude, high speed and hostile exterior environment, (e.g., high skin temperature) together with the requirement for world-wide operation all point to the choice of a self-contained navigation system operating independent of exterior aids. The accuracy of state of the art inertial navigation systems is adequate for the mission. Thus, the additional development costs expected for aided inertial systems - doppler - inertial or stellar inertial - can be avoided. The development of a Mach 7 doppler radar, the radome problem, and the star tracker window problem are avoided.

The radio and radar components specified for the recommended concepts have not been studied in detail in the Phase I study. These aids provide potential advantages and require further study during Phase II before a definite selection can be made. Radome and antenna heating is expected to be a critical problem and may have a decisive effect on the final configuration.

To date, all inertial navigation systems capable of meeting the desired accuracy of 1.85 km/hr. (1 N.M./hour) have been gimbaled platform systems. Recently, however, a great deal of work has been directed toward the development of strapdown inertial navigation systems. The strapdown system potentially offers the advantages of higher reliability, lower weight and power, and lower cost. Since the Euler angles must be computed in a strapdown system, more computer capacity is required than in the gimbaled system. For equal accuracy inertial components, the accuracy of the strapdown system is lower than that of the platform system due to gyro torquer inaccuracy and errors due to computer round-off and spped limitations. The potential advantages and problems with a strapdown inertial system will be examined as part of the advanced technology concept Phase II study.

World-wide coverage radio aids such as Omega or navigation satellites can only be included as part of an advanced concept because the operational date for these systems is uncertain. The launch vehicle system under study is not expected to implement these radio aids. However, if they have been implemented for other national requirements there may be cost, accuracy or reliability advantages in their use for updating the launch vehicle system.

The use of statistical data processing techniques (such as Kalman filtering) has not been examined in detail during Phase I because of the broad scope of the trade study. The accuracy advantage with statistical data processing represents an expected advantage with this technique which is currently under development in other applications. Additional work is required on error analysis and determination of computer requirements.

The guidance law approach recommended for the current technology concept is the current state of the art technique of dividing the flight profile into segments and defining a guidance law appropriate for each segment. The concept includes a flight profile generator for predicting the end conditions that will occur if the current definition of the guidance command profile is followed. Errors in the predicted end conditions result in a modification of the guidance command profile so that the rendezvous condition is satisfied.

The guidance law approach recommended for the advanced technology concept is the lambda matrix technique. The Phase II effort is to determine the feasibility and penalties associated with the lambda matrix technique when applied to the rendezvous problem. Other modern control theory approaches are to be considered also if initial studies of the lambda matrix technique indicates a need.

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APPENDIX A1

Payload Exchange Ratios

Payload Penalty Resulting from Stage 1 Fuel Penalty

Payload penalty can be considered in two ways for first stage errors

1) off-loading stage 2 propellant
2) reducing stage 2 size

The first method assumes a fixed vehicle design. The inert weight is fixed. Therefore a reduction in Stage 1 payload requires a reduction in Stage 2 propellant and payload to achieve the same orbit. The orbital or Stage 2 payload penalty is equivalent to a one-for-one reduction in permanent weight.

$$\Delta P = \frac{W_{SB}}{(W_{SB}/W_{EB})}$$

where

$$\frac{W_{SB}}{W_{EB}} = 1 + \text{EXP}\left(\frac{\Delta V_I}{I_s\ g}\right)$$

and

ΔV_I - ideal velocity gained along stage 2 nominal trajectory
I_s - propellant specific impulse
g - sea level gravitational constant (32.2 fps)
W_{SB} - stage 2 start burn weight
W_{EB} - stage 2 end burn weight
ΔP - payload penalty

For the current study, the ratio of start burn to end burn weight is about 4.66, so 4.66 pounds of first stage payload is equal to 1 pound of orbital payload.

$$\Delta P = .214 \quad W_{SB}$$

$$\Delta W_P = .786 \quad W_{SB}$$

where

ΔW_P - propellant penalty

The second method assumes a rubber stage 2 where the inert and propellant weights vary according to the structural relation

$$\frac{W_P}{W_P + W_I} = \lambda'$$

where:

W_P - stage 2 propellant weight

W_I - stage 2 inert weight

λ' - constant equal to .882

Combining this relation with the mass ratio

$$\frac{W_{SB}}{W_{EB}} = \frac{W_P + W_I + \rho}{W_I + \rho}$$

$$\Delta\rho = .109 \quad \Delta W_{SB}$$

$$\Delta W_I = .105 \quad \Delta W_{SB}$$

$$\Delta W_I = .214 \quad \Delta W_{SB}$$

$$\Delta W_P = .786 \quad \Delta W_{SB}$$

In this case 9.16 pounds of first stage payload is equivalent to 1 pound of orbital payload, 96½% lower penalty than offloading fuel for a fixed design. The design will be considered fluid for this study so the lower penalty relationships will hold.

Payload Penalty Resulting From Stage 2 Performance Penalty

Again considered the stage 2 design is fluid and start burn weight is constant, the stage 2 performance is defined by ideal velocity

$$V + \Delta V = I_s g \cdot \ln\left(\frac{W_{SB}}{W_{EB} + W_{EB}}\right) \qquad (1)$$

where:

V	=	ideal velocity
ΔV	=	ideal velocity penalty
I_s	=	specific impulse
g	=	gravitational constant
W_{SB}	=	start burn weight
W_{EB}	=	end burn weight
ΔW_{EB}	-	end burn weight penalty resulting from ΔV

The sensitivity of weight (i.e. propellant) to a change in ideal velocity is

$$\frac{dW}{dV} = \frac{-W}{I_s\ g} \qquad (2)$$

If the additional performance is added at the end of the trajectory, then

$$\left.\frac{dW}{dV}\right|_{end} = \frac{-W_{EB}}{I_s\ g} = -2.0 \text{ (nominal) } \frac{lb}{fps} \qquad (3)$$

Since, $W_{SB} = W_P = W_I + \ell$ (4)

where W_P = Propellant weight

W_I = Inert Weight

ℓ = Payload

and, $W_{EB} = W_I + \ell$ (5)

then, $\Delta W_{EB} = \Delta W_I + \Delta \ell = -2 \cdot \Delta V$ (6)

and, $\Delta W_{SB} = \Delta W_P + \Delta W_I + \Delta \ell = 0$ (7)

Therefore, $\Delta W_P = 2 \cdot \Delta V$ (8)

From the constant structural relation,

$$\lambda' = \frac{W_P}{W_I + W_P} = \frac{\Delta W_P}{\Delta W_I + \Delta W_P} \qquad (9)$$

The additional structural weight required is

$$\Delta W_I = \left(\frac{1 - \lambda'}{\lambda'}\right) \Delta W_P \qquad (10)$$

Combining (6) and (10),

$$\Delta \ell = \frac{dW}{dV}\left(\frac{\Delta V}{\lambda'}\right) = 2.27 \cdot \Delta V \qquad (11)$$

Therefore each 1 foot per second correction made at the end of the trajectory is equivalent to 2.27 pounds of orbital payload.

APPENDIX A2

Methods of Correcting Stage Timing Errors During First Stage Operation

This appendix describes four first stage flight techniques which allow for correcting stage timing errors accrued during first stage operation as a result of dispersions in aerodynamics, propulsion and atmosphere. The fuel penalty for each method is determined as a function of stage timing error correction capability and, shown in Figure A2-1. The negative penalty simply means the original choice of nominal path was not the minimum fuel path.

Lead Time Method

The lead time method is shown schematically in Figure A2-2 as a first stage ground track profile. The orbit plane intercept point is the place where staging starts when the minimum fuel utilization condition occurs. However to allow for time lag of the first stage, the nominal staging point is delayed with an associated fuel penalty.

$$\Delta F_N = K_1 \cdot \Delta R_{lag} \tag{1}$$

where:

ΔF_N = additional fuel required to fly the nominal

K_1 = pre-stage fuel usage per unit range

ΔR_{lag} = additional range to fly a nominal path which allows for predetermined 3σ lag time errors.

In addition to the fuel used to fly a suitable nominal path, correction fuel must be carried for nulling lead time errors.

$$\Delta F_c = K_1 \Delta R_{lead} \tag{2}$$

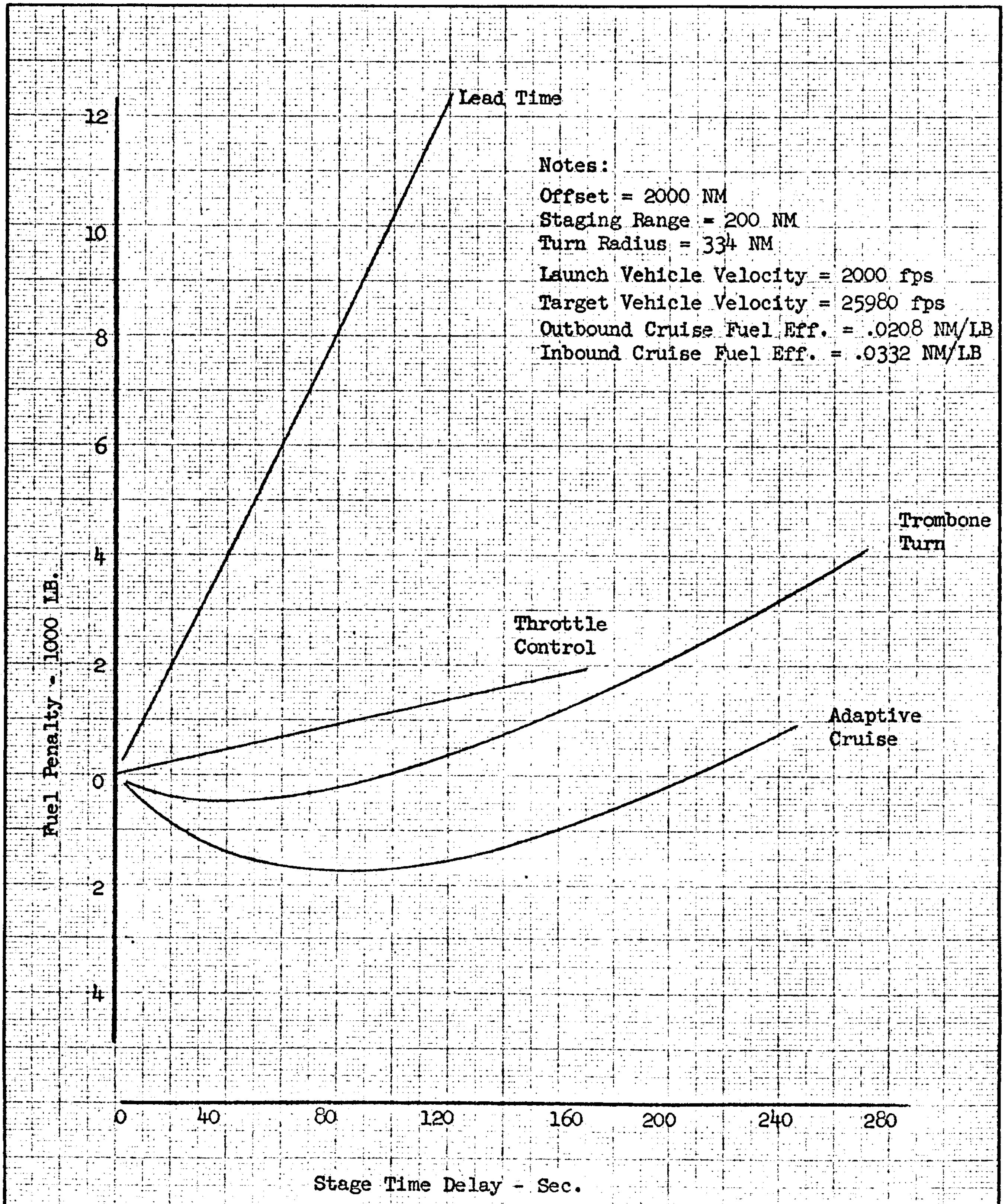
where:

ΔF_c = correction fuel carried to allow for lead time errors

ΔR_{lead} = additional range to allow for lead time errors

The requirement at the staging point to satisfy the rendezvous condition is that the launch vehicle and target satellite have a specified separation distance; this distance depends on the nominal second stage path during the thrust period and on the nominal transfer path. The difference in velocity between the launch vehicle and the satellite determines the change in desired staging time Δt_E when the launch vehicle flys a loiter distance ΔR . The exact relationship for a change in stage time, to obtain the required relative geometry is

$$\Delta t_E = \frac{\Delta R_1}{V_1} - \frac{\Delta R_2}{V_2} \tag{3}$$



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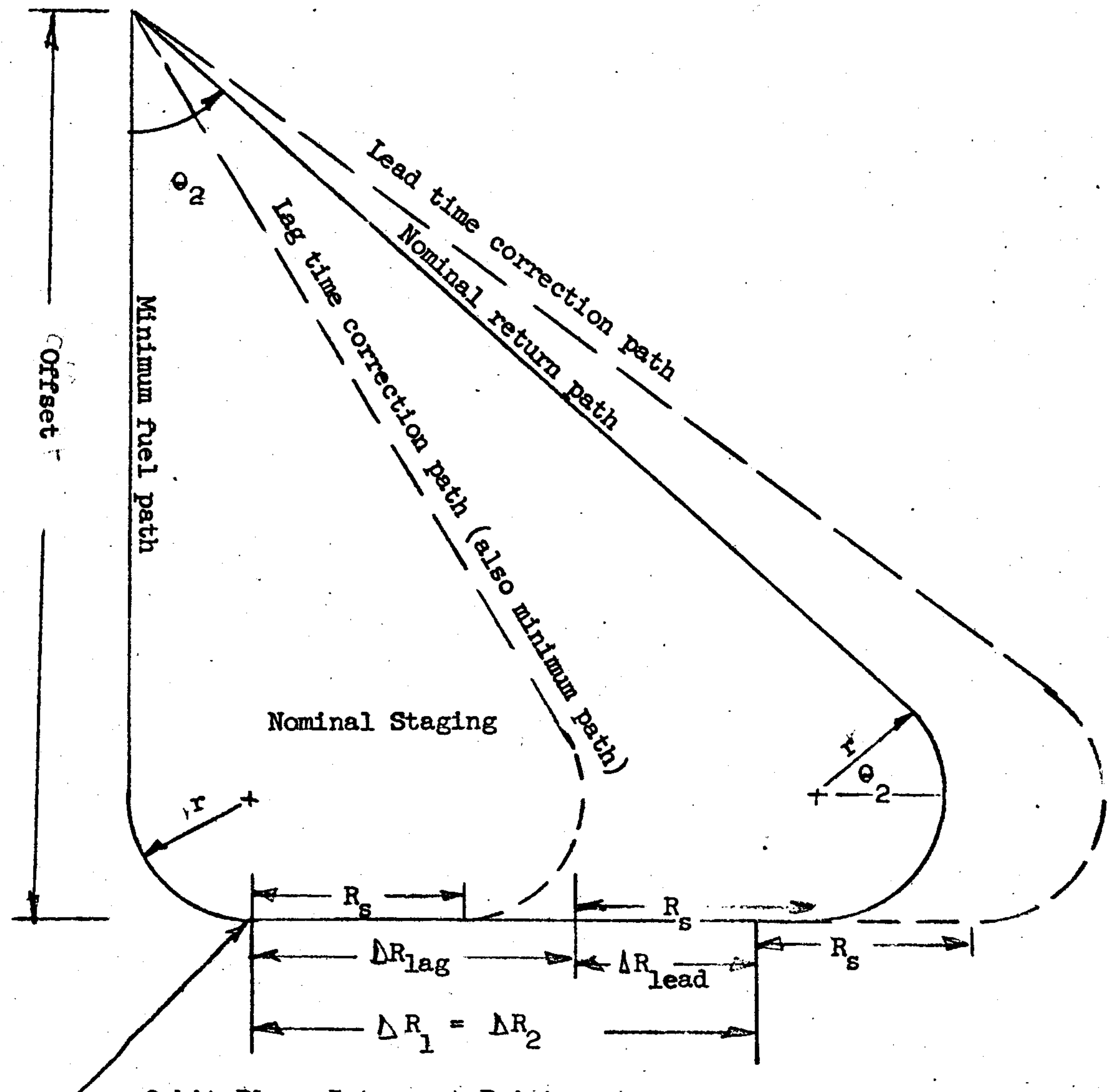

Notes:

ΔR_{lag} = Incremental range due to first stage lag time

ΔR_{lead} = Incremental range due first stage lead time

R_s = Minimum in-plane range to accomplish staging

ΔR_1, R_2 = Additional range caused by launch vehicle and target vehicle.

θ_2 = Return heading angle.

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BOEING NO. Figure A2-2

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where

Δt_E = stage time error

ΔR_1 = additional range covered by launch vehicle

ΔR_2 = additional range covered by target satellite

V_1 = launch vehicle velocity

V_2 = target satellite velocity

Note that ΔR_2 can be positive or negative depending on whether the staging point moves up-range or down-range. For the lead time method the first stage and target satellite fly the same path so,

$$\Delta R_1 = \Delta R_2 \tag{4}$$

and the staging point moves down range. Therefore,

$$\Delta R = \frac{V_1 V_2 \Delta t_E}{V_2 - V_1} \tag{5}$$

or

$$\Delta t_E = \frac{\Delta R (V_2 - V_1)}{V_1 V_2} \tag{5a}$$

Because of the stretched range on the outbound cruise, additional fuel is required for the return cruise,

$$\Delta F_2 = K_2 \Delta R \tag{6}$$

The additional range is defined by the geometry of figure 1.

$$\Delta R_r = \frac{R_s + r(1+\cos\theta_r) + \Delta R_2}{\sin\theta_r} - \frac{R_s + r(1+\cos\theta_2')}{\sin\theta_2'} \tag{7}$$

where:

ΔR_r = additional return cruise range due to nominal flight path or/and correction of phasing errors

R_s = minimum staging range

r = airbreather cruise course turn radius

θ_2 = return heading relative to minimum fuel outbound cruise heading

θ_2' = minimum fuel return heading angle

K_2 = post-stage fuel usage per unit range

ΔR_2 = change in range along the orbit plane resulting from outbound cruise path other then the minimum fuel path (from equation (5) for lead time method)

It will be shown that ΔR_2 can be positive as in the lead time method thus increasing the return range and associated fuel penalty or it can be negative resulting in a shorter return range and providing a compensating effect on the outbound fuel penalty.

In consideration of the outbound and inbound cruise penalties, the resultant fuel penalty is

$$\Delta F = \Delta F_1 + \Delta F_2 \tag{8}$$

where

ΔF = first stage fuel penalty resulting from lead and/or lag time correction capability

ΔF_1 = prestage fuel penalty

ΔF_2 = post-stage fuel penalty

Trombone Turn Method

This method is shown schematically in Figure A2-3 as a first stage ground track profile. In this case the time variation occurs during an S-turn for greater or lesser heading angle change depending upon whether the first stage leads or lags the target vehicle. Staging occurs at the orbit plane intercept point. Equations (1) and (2) apply for determination of the fuel penalty but the additional range of stage 1 is defined by the relation:

$$\Delta R_1 = r(\theta - \sin\theta) \tag{9}$$

where:

ΔR_1 = incremental range covered by the launch vehicle

θ = heading change in the turn

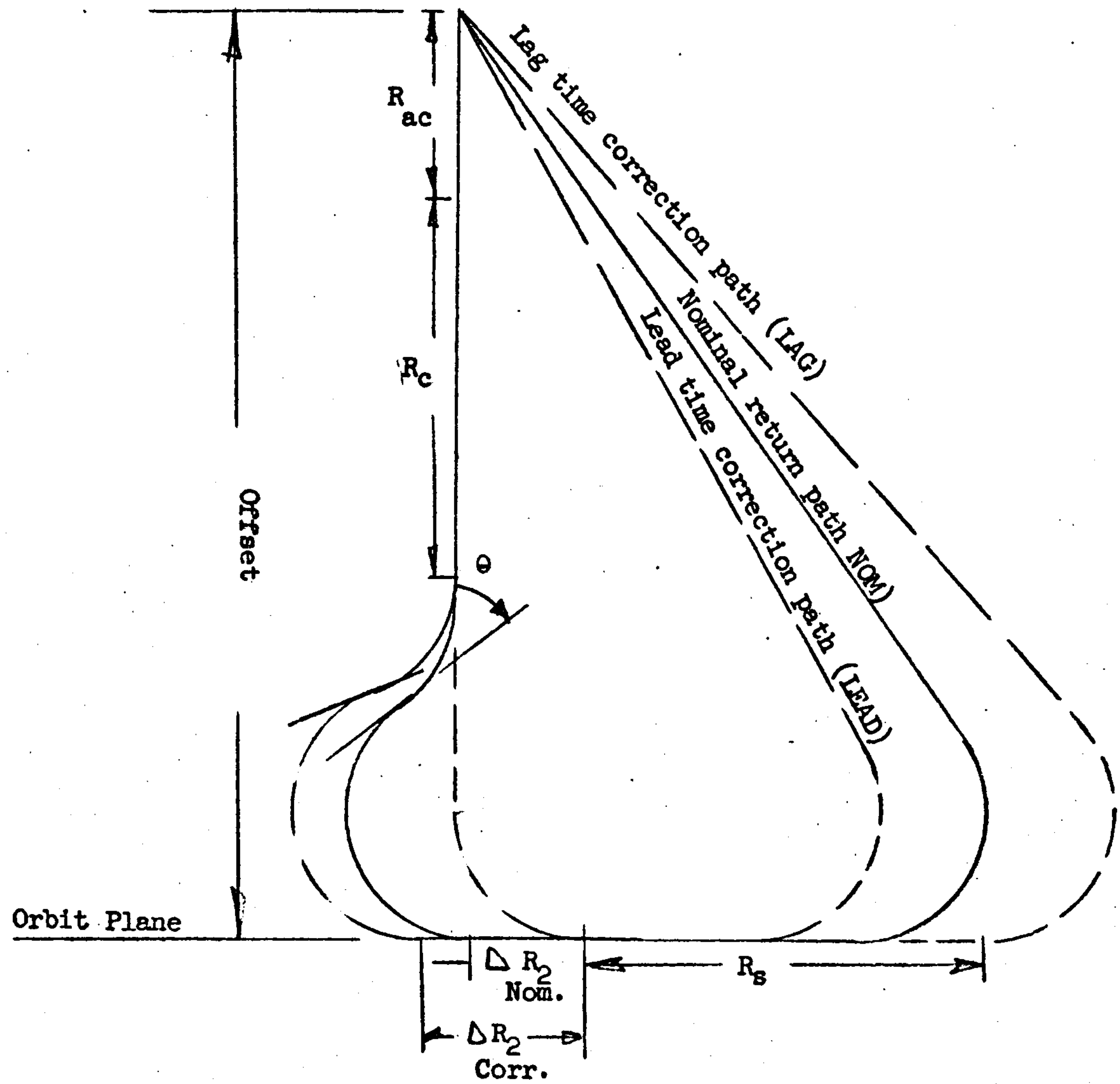
and the additional range of the target vehicle is

$$\Delta R_2 = -2r(1 - \cos\theta) \tag{10}$$

Substituting the relationships of equations (9) and (10) into equation (3),

$$\Delta t_E = \frac{r(\theta - \sin\theta)}{V_1} + \frac{2r(1-\cos\theta)}{V_2} \tag{11}$$

It can be seen by comparing equations (5a) and (11) that the trombone method provides a better time correction technique than the lead time method. This is because the lead time method depends on the difference in velocity to kill time where as the trombone method has the total target vehicle velocity plus some varying component of the launch vehicle velocity working for it. In other words stage time delays are



Notes:

ΔR_2 = Target Vehicle Range Change
R_s = Staging Range Allowance
Θ = Heading Angle Change
R_{ac} = Acceleration Range
R_c = Cruise range before heading change

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created more efficiently with the launch vehicle in any direction other than the direction of the target vehicle. If the heading change angle were allowed to increase to 90 degrees, the launch vehicle velocity would add directly to target vehicle velocity for purposes of killing time. Equation (5) would become $\Delta R = V_1 V_2 \Delta t_E / (V_1 + V_2)$ for that period of time which the launch vehicle flies in the opposite direction as the target vehicle. For some ranges of heading angle change, another favorable influence is the effect of staging point shift on return range. Equation (7) shows that for a negative shift in staging point the return range is reduced.

$$\Delta R_r = \frac{R_s + r(1+\cos\theta_r) - 2(1-\cos\theta)}{\sin\theta_r} - \frac{R_s + r(1+\cos\theta_r')}{\sin\theta_r'} + r(\theta_2 - \theta_2') \tag{12}$$

where:

θ_2 = return heading angle relative to the minimum fuel outbound heading angle

θ_2' = return heading angle for the minimum fuel return path relative to the minimum fuel outbound heading angle

This is the case for the nominal flight path using the trombone method and has a compensating effect on the fuel penalty. However, for θ_2 less than zero or greater than θ_r the effect becomes additive. Preliminary analysis indicates that for expected three sigma stage time errors, the effect is always compensating for flying the nominal but adverse for lag time error as indicated by Figure A2-3.

Adaptive Cruise Method

This method is shown schematically in Figure A2-4 as a launch vehicle ground track. It employs a heading angle change followed by a period of cruise then a final turn into the target plane. Staging occurs at orbit plane intercept. It is mathematically similar to the trombone turn method, but equations (9) and (10) have the added term resulting from launch vehicle cruise following the heading change.

$$\Delta R_1 = r(\theta - \sin\theta) + R_c'(1-\cos\theta) \tag{13}$$

and

$$\Delta R_2 = -2r(1-\cos\theta) - R_c' \sin\theta \tag{14}$$

where

R_c' = cruise range following the heading angle change

It is determined from the total offset, the acceleration range and the cruise range before the heading angle change as a function of the heading angle change

$$R_c' = \frac{\phi_{ff} - R_{ac} - R_c - r(1 + 2\sin\theta)}{\cos\theta} \tag{15}$$

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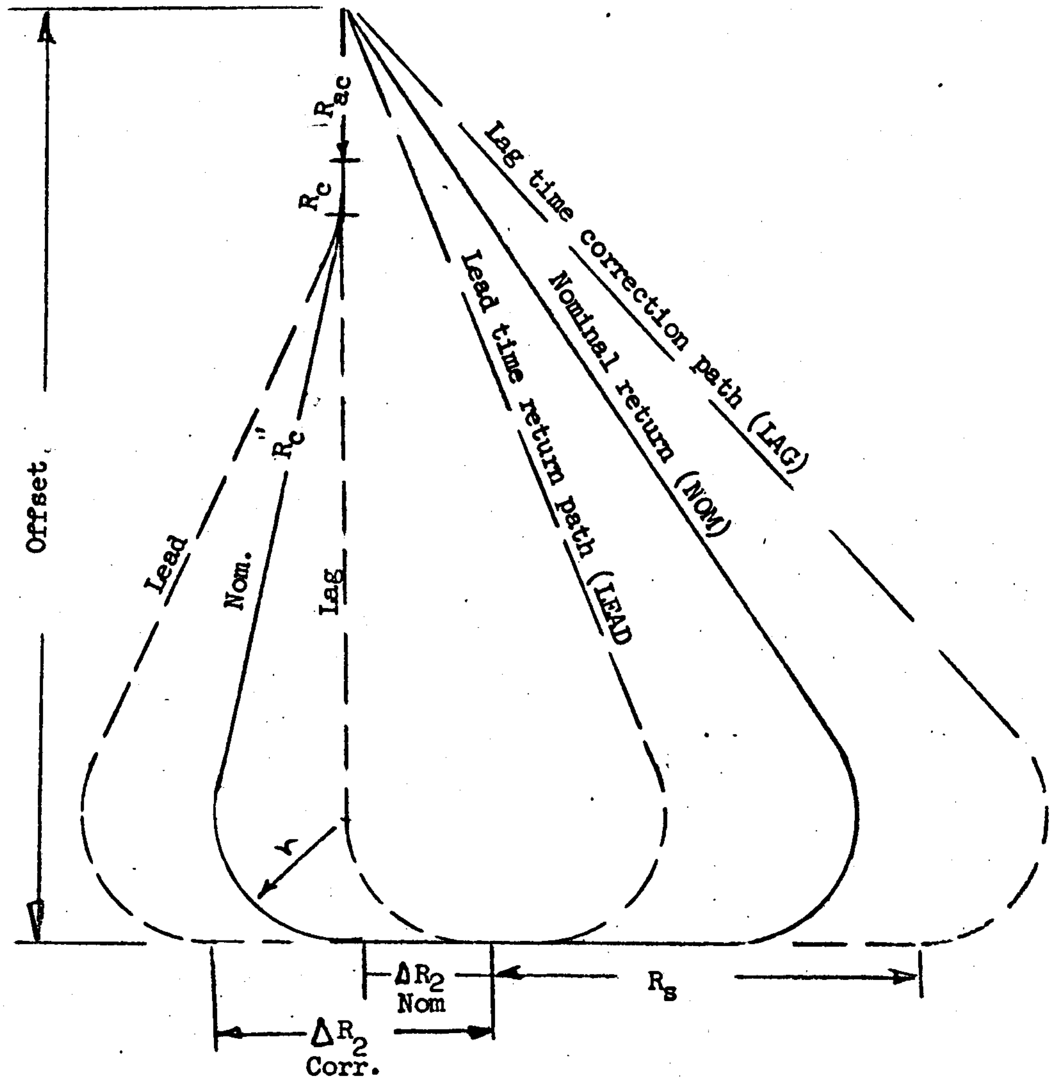

Notes:

ΔR_2 = Target Vehicle Range Change

R_s = Staging range allowance

θ = Heading angle change

R_{ac} = Acceleration range

R_c = Cruise range before heading change

R_c' = Cruise range after heading change

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where

ϕ_{ff} = orbit offset

R_{ac} = launch vehicle acceleration range

R_c = cruise range prior to the heading change

The uprange term R_c (1 - Cos θ) in the ΔR_1 equation has the effect of reducing the heading change required to correct a given error. This can be a significant factor in determining of effect of where along the launch vehicle flight path the stage time errors become known. If they occur early in the trajectory, the adaptive cruise method becomes attractive because of the smaller deviation from the minimum fuel flight path. However if large errors occur in the cruise phase the trombone method may well be the more economical scheme. Preliminary analysis indicates large errors are possible in the acceleration phase. For example, stage time errors are quite sensitive to deviations in thrust during acceleration. On the other hand, winds have a dominating influence during the cruise/turn. In practice, a combination of all three time correction methods would probably be used because no greater guidance complexity is added to the system by the addition of these simple equations than is required to determine the position of the moving target plane under nominal flight path conditions. If indeed the combination method were used then acceleration phase errors as well as early cruise errors could be nulled by the adaptive cruise method, cruise time errors could be nulled by the trombone turn method and final phasing errors could be corrected by the lead time method. This allows the large stage time errors to be corrected by the most economical method.

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Throttle Control

Timing errors can also be corrected by throttle control. The fuel penalty is defined by the partial derivative

$$\Delta F = \frac{\partial F}{\partial t} \Delta t \qquad (1)$$

where: Δt is the timing error

$\frac{\partial F}{\partial t}$ = rate of fuel usage with respect to timing error

ΔF is the fuel penalty

The derivative is not conveniently evaluated therefore it is rewritten

$$\frac{\partial F}{\partial t} = \frac{\partial F / \partial V}{\partial t / \partial V} \qquad (2)$$

where: V is the cruise velocity

The denominator can be evaluated since

$$t = \frac{R}{V}$$

where: R is the range to go to staging
V is the cruise velocity

then

$$\left.\frac{dt}{dV}\right|_{R=\text{Const.}} = -\frac{R}{V^2} \qquad (3)$$

To evaluate the numerator an expression must be written for fuel in terms of velocity. From the range equation.

$$F = W\left(1 - \text{EXP}\frac{-R}{RF}\right) \qquad (4)$$

where
F = fuel
W = weight when timing error is recognized
R = range to go
RF = range factor

which can be differentiated with respect to velocity

$$\left.\frac{\partial F}{\partial V}\right|_{R=\text{Const}} = -W\left(\text{EXP}\frac{-R}{RF}\right)\left(\frac{R}{RF^2}\right)\left(\frac{\partial RF}{\partial V}\right) \qquad (5)$$

but

$$RF = \frac{(L/D)V}{sfc} \qquad (6)$$

where (L/D) = lift to drag ratio
V = cruise velocity
sfc = specific fuel consumption

Both L/D and sfc are velocity dependent therefore;

$$\frac{\partial RF}{\partial V} = \frac{sfc\left(L/D + V\frac{\partial (L/D)}{\partial V}\right) - \frac{L}{D}V\frac{\partial sfc}{\partial V}}{sfc^2} \tag{7}$$

This derivative evaluated at cruise velocity is

$$\left.\frac{\partial RF}{\partial V}\right|_{V=7000 FPS} = 0.58 \frac{NM}{fps} \tag{8}$$

Substituting equation (8) into equation (5) and evaluating at the start of cruise

$$\left.\frac{\partial F}{\partial V}\right|_{\substack{\text{start}\\ \text{cruise}}} = -2.6 \frac{lb}{fps}$$

and evaluating equation (3) at start cruise

$$\left.\frac{\partial t}{\partial V}\right|_{\substack{\text{start}\\ \text{cruise}}} = -0.225 \frac{sec}{fps}$$

For a 60 second increment in time

$$\Delta V_{CR} = -267 \text{ fps}$$

$$\Delta F = 693 \text{ lb}$$

Since the velocity is constrained to 7000 fps, the penalty for correcting ± 60 seconds would be 1386 lb.; 693 to fly the nominal and 693 to correct an additional 60 seconds.

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APPENDIX A3

Terminal Phase Sensor Weight Trades

Reference

1. J. D. Mallett and L. E. Brennan, "Cumulative Probability of Detection for Targets Approaching a Uniformly Scanning Search Radar", Proceedings of the IEEE, April 1963, pp 596-601.

Weight penalties in the rendezvous terminal phase to correct for navigation and guidance errors in the preceeding flight phases are caused by two factors: (1) the weight penalty in performing the terminal maneuver, and (2) the weight penalty associated with the target acquisition and tracking sensor. The sensor weight penalties are significant for the non-cooperative target situation. A non-cooperative target may occur for a rescue mission, a transponder malfunction in the friendly target satellite, or a military rendezvous mission. The acquisition and tracking sensor penalties for a cooperative target are a fixed value independent of the navigation and guidance errors for the designs used in Gemini and Apollo. In these applications the cooperative target seeker range is very much greater than the expected errors.

The approach is developed below for obtaining target seeker weight trades versus errors for the non-cooperative target satellite. An analysis of the detection range performance of a radar target seeker is summarized here from Reference 1. A radar sensor is the conventional solution to the rendezvous terminal phase sensor problem in the current state-of-the-art applications; thus, the current study will be limited to the radar case.

The starting point for considering the radar detection range performance R is to normalize in terms of R_o, the range for unity signal-to-noise ratio.

$$R_o^4 = \frac{\bar{P} G A_e \sigma T_d}{(4\pi)^2 K T_e L}$$

where

- $\bar{P}$ = average transmitted power
- G = transmitting antenna gain
- A_e = effective receiving antenna area
- σ = target echo area
- T_d = dwell time, the time the target is within the radar beam during a single scan
- T_e = effective receiver temperature describing its noise level
- L = loss factor for various component attenuations

This equation assumes that the radar is designed for coherent integration for the time T_d or equivalently that the signal is passed through a matched filter.

The cumulative detection probability P_c is the probability that an approaching target will have been detected at least once by the time it reaches a given range R_1. The distance the target travels relative to the radar in one search frame time, T, $\Delta = V_c T$ where V_c is the target closing velocity. The reference shows that there is an optimum valve of the ratio Δ / R_1 for a given cumulative probability and target cross section fluctuation model and equation (1) is rewritten as

$$R_1^3 = \frac{R_o^4}{\Delta} \cdot \frac{\bar{P} A_e \sigma}{4\pi K T_e L \Omega V_c} \cdot Q_i^3 \qquad (2)$$

where Ω is solid angle of the search frame and Q_i is a correction factor for non-coherent integration. Relationships used in obtaining (2) from (1) are $\omega = \frac{4\pi}{G}$ = the solid angle of the radar beam, $\frac{T_e}{T} = \frac{\omega}{\Omega}$ and $\Delta = V_c T$

Equation (2) for the optimum Δ / R_1 ratio indicates that the cumulative probability detection range depends on the cube root of the uncertainty solid angle and the closing velocity.

$$R_1 = \frac{K}{\Omega^{1/3} V_c^{1/3}} \qquad (3)$$

where K has a constant value for a radar with a given weight and electrical power input.

The factors determining the radar weight for given detection range, search angle, and closing velocity are the antenna size A_e the average transmitted power $\bar{P}$ receiver sensitivity T_e, losses L and target size σ.

The radar weight W can be describe empirically as a function of the range R, for a given search solid angle and closing velocity:

$$W = W_o + B R_{10} + C R_{10}^2 = 54 + 1.6 R_{10} + .02 R_{10}^2 \qquad (4)$$

is the relationship that has been used in a previous terminal phase sensor trade study, where W is in pounds and R_{10} in nautical miles. The electrical power required will be assumed to be proportional to the radar weight. A 150 lb radar requires 1200 watts input power. The 150 lb radar has a 40 NM range at a 2000 fps closing velocity and 40° search solid angle.

The target seeker will be used intermittantly during the mission increasing the peak load requirements on the electrical power subsystem. A rendezvous vehicle designed with a fuel cell power subsystem had a 55 lb increase in weight to handle the 1200 watt increase in peak load associated with the radar sensor. The short time, 15 minutes, for operation of the radar sensor required an insignificant (1 lb) increase in fuel weight.

The empirical formula (4) has been checked with a 1962 preliminary design of a rendezvous radar and found to be optimistic by 10%. This check is considered good for the trade study application in the current study.

Using equation (3) to modify equation (4) to a 400 fps closing velocity condition gives

$$W = 54 + 0.935\, R_{10} + 0.0067\, R_{10}^2$$

If A is the angle on one side of a square search pattern, $\Omega = A^2$ and using equation (3) gives W as a function of the range R, and search angle A.

$$W = 54 + 0.935\left(\frac{A}{40}\right)^{2/3} R_1 + 0.0067\left(\frac{A}{40}\right)^{4/3} R_1^2 \qquad (5)$$

If E is the transverse error due to navigation and guidance in the direction perpendicular to the nominal closing velocity direction, the half search angle is given by

$$\sin\left(\frac{A}{2}\right) = \frac{E}{R_1}$$

Thus, the radar weight W is a function of E and R1. For a given error E there is an optimum scan angle established by the condition.

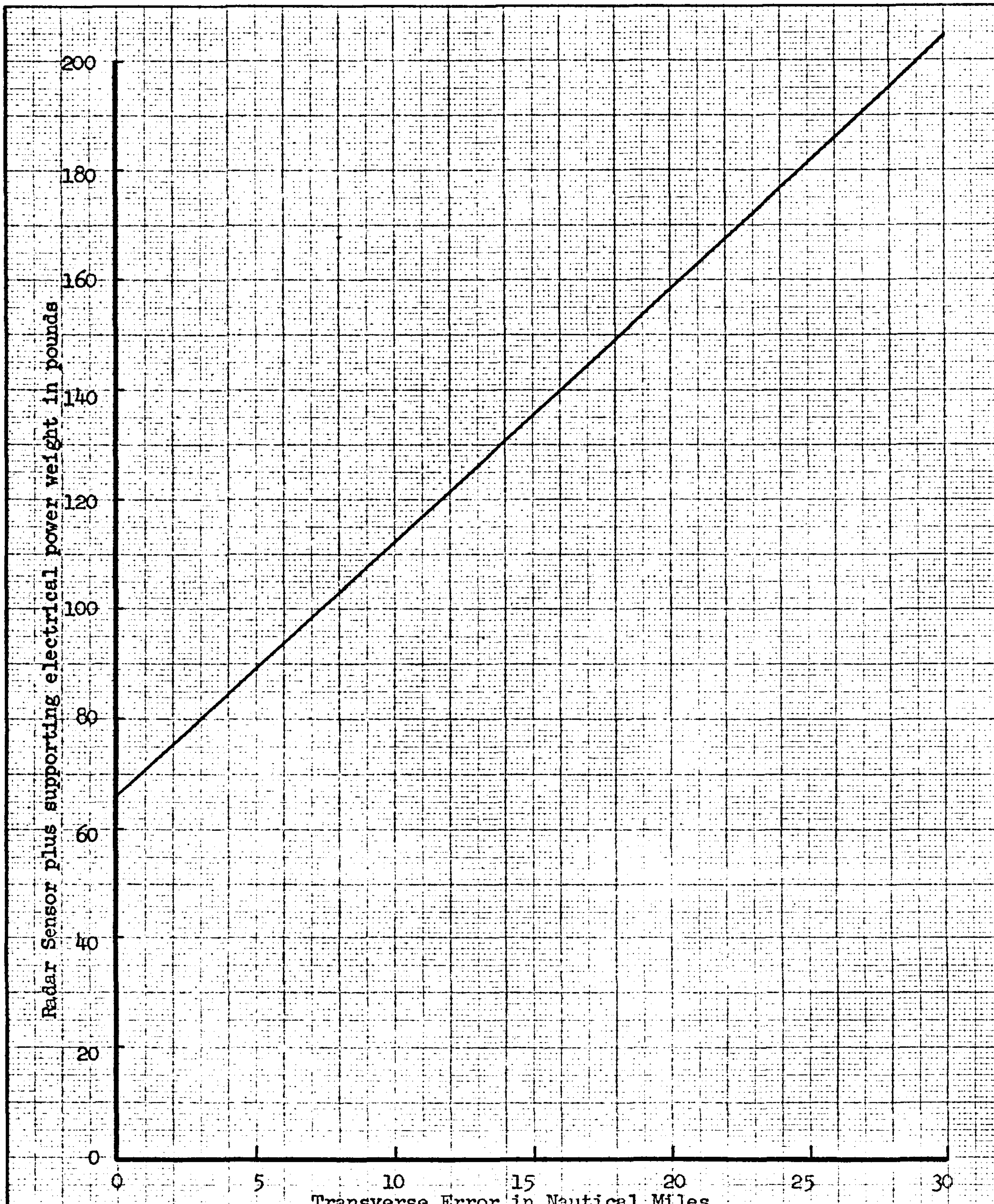
$$\frac{\partial W}{\partial A} = 0$$

The analytical expression obtained is complex. Using equations (5) and (6) the minimum weight sensor was determined by a numerical search process for two assumptions of transverse error, 10 MN and 25 NM. The resulting sensor characteristics are given in the following table:

TABLE 1

RADAR CHARACTERISTICS, MINIMUM WEIGHT	TRANSVERSE ERROR 10 NM	25 NM
Detection range	13.1 NM	30.5 NM
Scan angle	100° x 100°	110° x 110°
Radar weight	81 lbs.	132 lbs.
Supporting Power Weight	30 lbs.	49 lbs.
Total weight increase	111 lbs.	181 lbs.

These sensor characteristics were obtained from the original sensor weight model, equation (4). A detailed radar sensor preliminary design would be required to verify the characteristics in Table 1. The effort to accomplish this is beyond the scope of the current trade study. The sensor weight trend that has been obtained in Table 1 is used to give the approximate trade of sensor weight plus supporting electrical power versus navigation and guidance transverse error. This trade is given in Figure A5-1. A certain minimum weight is required to provide an error sensing sensor when the transverse errors are small.



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REFERENCES - DATA SOURCES

A. D2-100255, "Reliability Analyses and Supporting Data - Lunar Orbiter": The Boeing Company, June 17, 1964.

B. D2-82665-1, "Burner II Reliability Analyses Report": The Boeing Company, January 7, 1966.

C. T5-607/3061, "Astrionics Reference System Description" (Proposal for Voyager '69 Guidance and Control): Autonetics, Division of North American Aviation, Inc., March, 1965.

D. Autonetics Letter, 64AN/NSD2506, "Self-Contained Navigation Systems, Boeing Supersonic Transport": May 1, 1964.

E. R.-ED24181, "Proposal for Satellite Injection Guidance System": Honeywell Aero, June 18, 1964.

F. Engineering Report: "Reliability Study on Nortronics Subminiature Self Test Rate Gyro GR-H 4-T": March, 1963.

G. Report No. ATL-D-1072, "Infrared Horizon Sensors for Precision Attitude Measurement": Advanced Technology Laboratories, July 1, 1963.

H. SP-3802-64-2, "Reliability Estimate and Analysis Report for Mariner C. Vehicles 6931, 6932 (Agena):" Lockheed Missiles and Space Company, May 28, 1964.

I. Volume F (Reliability Autonetics Proposal to The Boeing Company for the Voyager Attitude Reference and Autopilot Subsystem): Autonetics - A Division of North American Aviation, July 5, 1965.

J. IBM Report No. 63-384-004, "Reliability Analysis and Estimate Report for Gemini Inertial Guidance System"; Contract No. POY 20163: IBM Federal Systems Division, July 15, 1963.

K. AFFDL-TR-64-168, Part II, "Research and Investigation on Satellite Attitude Control: K. C. Nichol; by General Electric Company for the Air Force Systems Command.

L. SGC Report No. 111R-6, "Revised Reliability Analysis of the ABLESTAR Stage": Space General Corp., February 27, 1963.

M. Report No. 63SD4257, "Stabilization and Control Subsystem Orbiting Astronomical Observatory Eighth Quarterly Report": F. W. Crimp; General Electric Company.

N. ASD-TDR-62-219 (Confidential), "Compilation and Analysis of Component Failure Rate Data (Missile and Spacecraft)": Planning Research Corporation.

O. FARADA Reports: Bureau of Naval Weapons.

P. D2-82724-3 (formerly D2-23834-1), "Reliability Analysis and Prediction Standards - Voyager Program": The Boeing Company, July, 1965.

Q. Report No. 64SD4246, "Report on the Nimbus Stabilization and Control Subsystem," B. D. Hatch; General Electric Company.

R. Report No. ASD-TDR-63-293, "Application of Low-Level Redundancy for Guidance System Reliability," Light Military Electronics, Dept., General Electric Company, March 1963.

S. D5-15611-1, Vol. 7, Book 1, "Saturn V Failure Effects Analysis and Criticality Determination Report, Instrument Unit Stage SA501 Launch Vehicle," The Boeing Company.

U. Data sources are classified.

COMPARISON OF FAILURE RATE DATA

Development Status and Source / System	FAILURE RATE IN FAILURE PER 10^6 HOURS					
	Existing	Source	Under Development	Source	Proposed	Source
Guidance and Control	1241.0	B	373.0	S		
Guidance and Navigation	7520.0	U			441.0	D
	1745.0	U	833.0	U	333.0	D
	1830.0	U			1733.0	U
	1740.0	J			921.0	U
					265.0	U
					304.0	U
					2620.0	U
					875.0	U
					750.0	U
					13.0	U
					700.0	U
					333.0	U

See Appendix A for Source Identification

COMPARISON OF FAILURE RATE DATA

Development Status and Source / Subsystem	FAILURE RATE IN FAILURE PER 10^6 HOURS: Existing	Source	Under Development	Source	Proposed	Source
Programmer	71	A				
	51	B				
Attitude Control	244	A				
Gyro Reference Assembly	44	Q				
	80	L				
Horizon Attitude Computer	150	Q				
	700	R				
Guidance, Navigation and Sequencing			21.0	S		
Control Subsystem			108.0	S		
Controls and Displays					350.0	U
					134.0	U
					432.0	U
					144.0	U
					123.0	U
					30.0	U

See Appendix A for Source Identification

COMPARISON OF FAILURE RATE DATA

Development Status and Source / Subsystem	FAILURE RATE IN FAILURE PER 10^6 HOURS: Existing	Source	Under Development	Source	Proposed	Source
Inertial Platform and Electronics	4560.0	U	455.0	U	806.0	U
	1840.0	U	2000.0	U	317.0	U
					136.0	U
					962.0	U
					320.0	U
					275.0	U
					192.0	U
					50.0	U
					144.0	U
Vertical Gyro System					18.0	U
Doppler Radar	6666.0	U	3333.0	U	348.0	U
					109.0	U
					260.0	U
					425.0	U
					142.0	U
					122.0	U

See Appendix A for Source Identification

COMPARISON OF FAILURE RATE DATA

Development Status and Source / Subsystem	FAILURE RATE IN FAILURE PER 10^6 HOURS: Existing	Source	Under Development	Source	Proposed	Source
Inertial Reference Unit	152.0	A			2700.0	U
	437.0	B			625.0	U
	553.0	H				
	432.0*	H				
	1351.0	J	244.0	S		
Horizontal Sensor	270.0	H				
	1700.0*	H				
Velocity Meter	903.0	H				
	2070.0*	H				
	119.0	B				
Flight Control Electronics	232.0	H				
	293.0	H				
	179.0	B				
Inertial Platform	847.0	J	190.0	S	70.0	D
					53.0	D
Stable Platform Elec.			28.0	S	226.0	D
					165.0	D
Computer	391.0	J	108.0	S	53.0	D
	537.0	U	16.0	S	39.0	D
					289.0	U
					255.0	
					128.0	
					196.0	
					117.0	
					806.0	
					268.0	
					54.0	
					63.0	
					374.0	U

NOTE: Data with * are based on field data, all others are based primarily on generic part data.

COMPARISON OF FAILURE RATE DATA

Component (Development Status and Source)	FAILURE RATE IN FAILURE PER 10^6 HOURS: Existing	Source	Under Development	Source	Proposed	Source
Gyro	40.0	A, H			1.0	C, D, I
	100.0	B			10.0	E
	147.8	F			20.0	I
	60.0	K			40.0	
	5.0	M				
	10.0	L				
	33.0*	N				
	300.0*	O				
	700.0*	O				
Accelerometer	20.0	B			1.0	I, P
	10.0	A			4.0	E
	12.2	H			100.0	P
Star Tracker	6.2	A, P			5.8	C
					6.8	P
					6.5	P
					21.0	
					8.0	
					33.0	
					117.0	
					78.0	
					8.7	
					50.0	
					12.0	
Sun Sensor	6.3	A			1.0	C
					2.2	P
					10.0	P
Horizon Sensor	17.2	G				
	28.0	G				
Astrocompass	5000.0	U				
	2630.0	U				

NOTE: Data with * are based on field data, all others are primarily based on generic part data.